ADVANCED CONCEPTS

Lynx Mission Concept Study

Advanced Concepts Office

April 7, 2017

National Aeronautics and Space Administration
www.nasa.gov

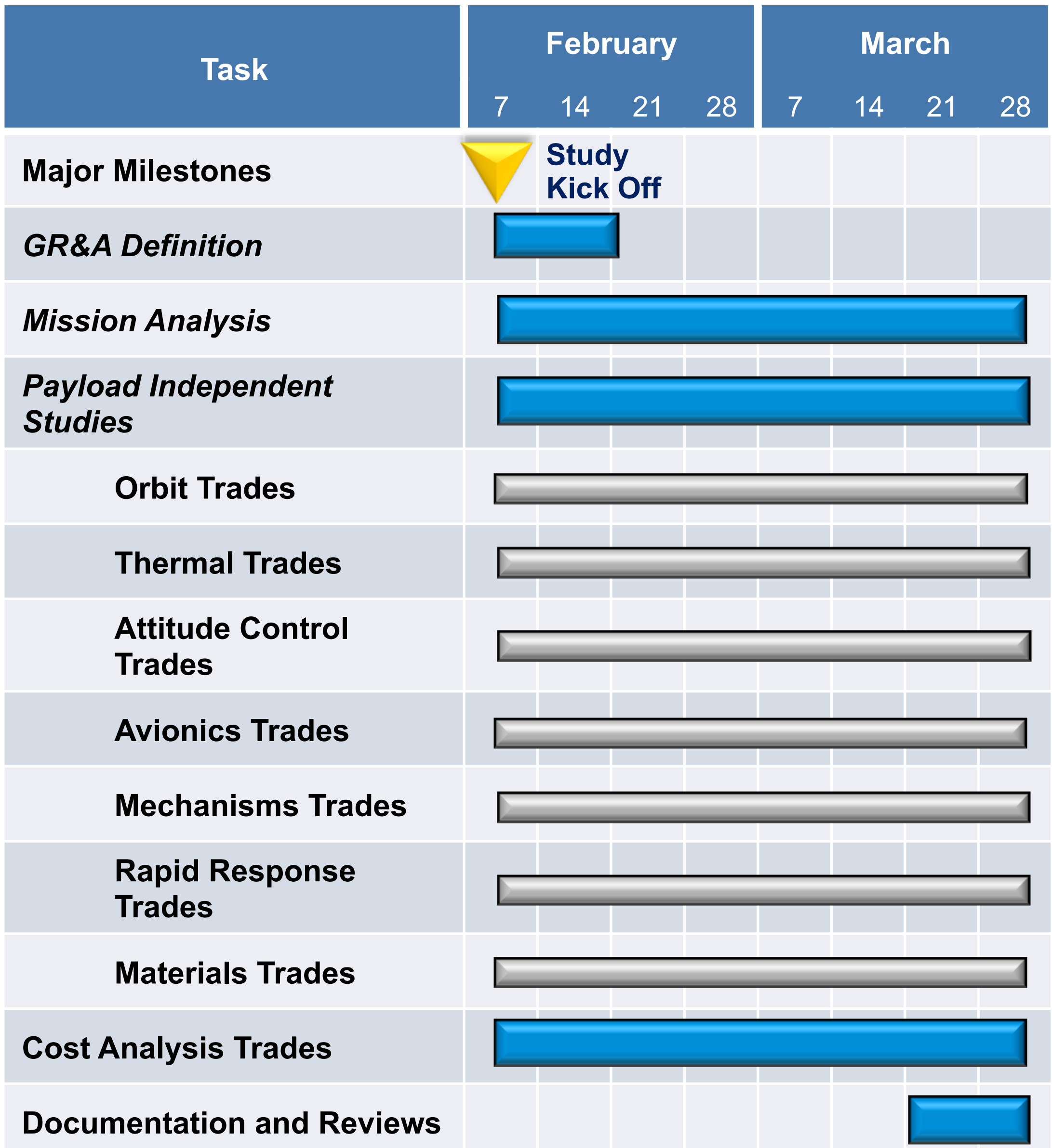
Agenda

- Overview
- Subsystem Reports
  - **Mission Analysis – Randy Hopkins**
  - **Environments –Rob Suggs**
  - **Structures – Jay Garcia**
  - **Thermal – Steven Sutherlin**
  - **Propulsion – Tyrone Boswell**
  - **GNC – Alex Dominguez**
  - **Avionics – Pete Capizzo**
  - **Mechanisms – Justin Rowe**
  - **Power – Leo Fabisinski**

Study Team

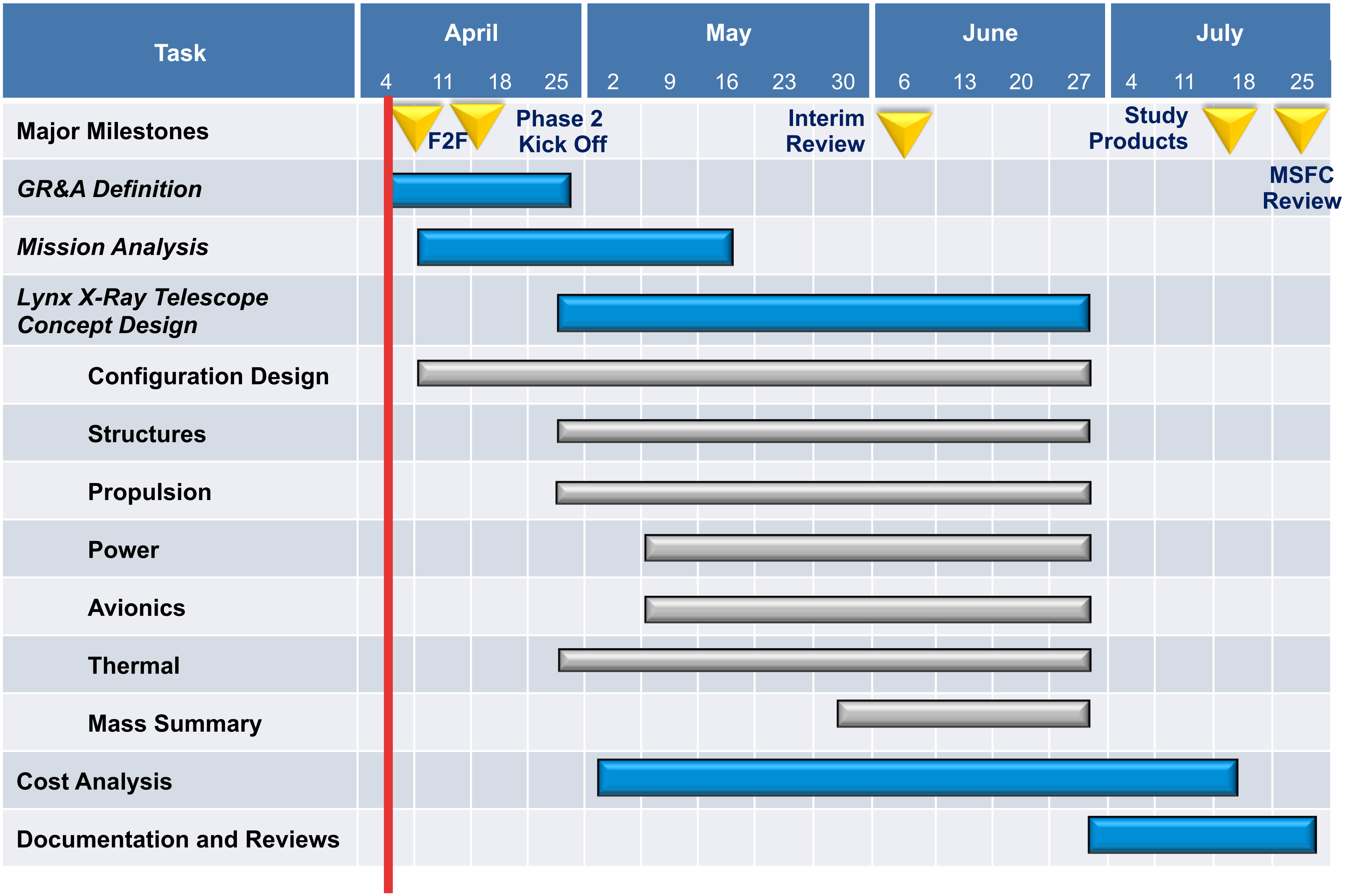
Role	Name	Organization
Principle Investigator Co-Principle Investigator	Jessica Gaskin Doug Swartz	MSFC / ZP12 MSFC / ZP12
Systems Manager	Karen Gelmis	MSFC / ZP21
ACO Team Lead	Jack Mulqueen	MSFC/ED04
ACO Study Lead	Andrew Schnell	MSFC / ED04
Mission Analysis	Randy Hopkins	MSFC / ED04
System Analysis	Mitchell Rodriguez	MSFC / ED04
Environments	Rob Suggs Emily Willis Michael Goodman	MSFC/EV44 MSFC/EV44 MSFC/EV44
Environments Design	Jim Howard Ian Small	MSFC/ES43 MSFC/ES43
Design & Configuration	Mike Baysinger	MSFC / ED04
Structures	Jay Garcia	MSFC / ED04
Propulsion	Tyrone Boswell	MSFC / ER23
Power	Leo Fabisinski	MSFC / ED04
Avionics	Pete Capizzo	MSFC / ES36
Thermal	Steve Sutherlin	MSFC / ED04
Mechanisms	Justin Rowe	MSFC/ED04
GNC	Alex Dominguez	MSFC/EV41
Cost Analysis	Spencer Hill Robbie Holcombe	MSFC / CS50 MSFC / CS50

Phase 1 Schedule



Phase 1 focuses on payload independent tasks

Phase 2 Schedule



Mission Analysis
Payload Independent Trades

Randy Hopkins

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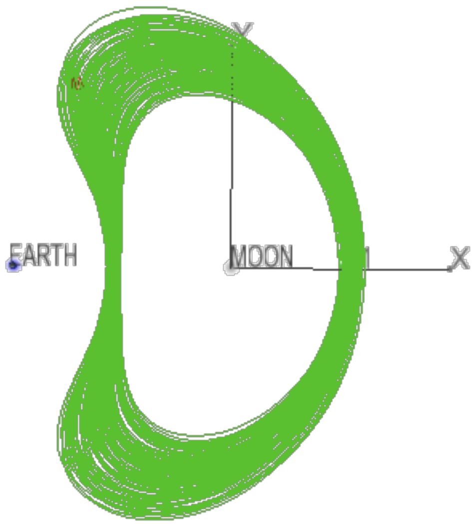
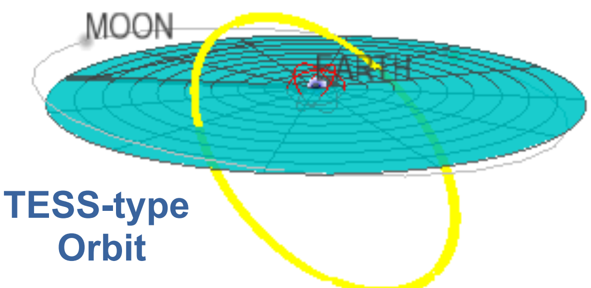
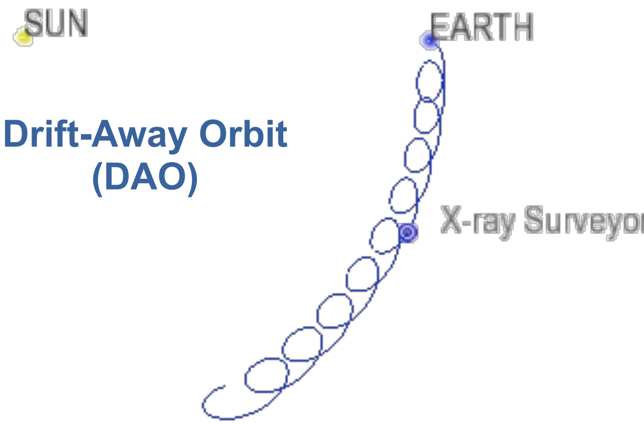
Orbit Trades: Orbits Considered



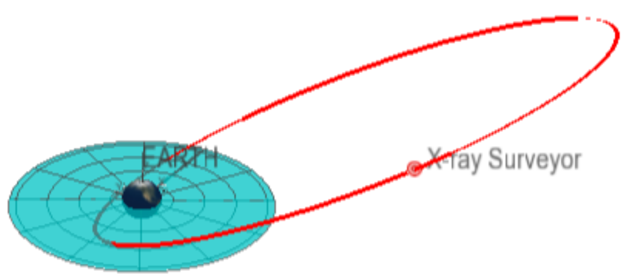
Lunar Distant Retrograde Orbit (LDRO)



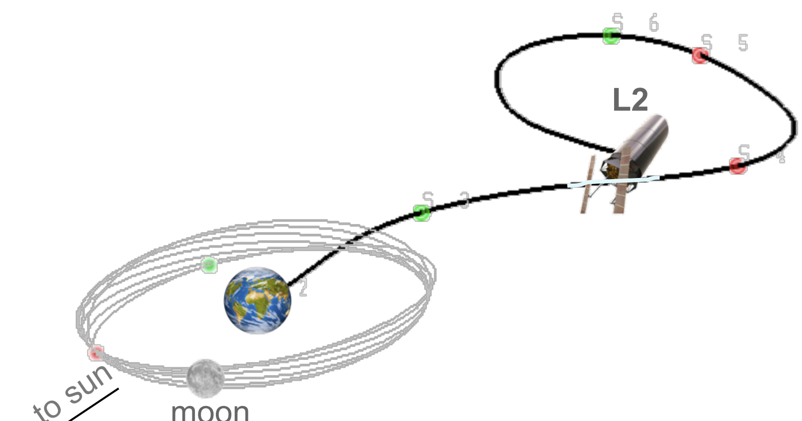
TESS-type Orbit



Drift-Away Orbit (DAO)



Chandra-type orbit (CTO)



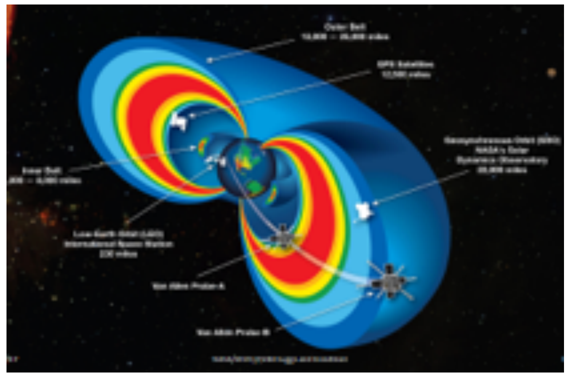
Sun-Earth L2

Orbit Trades: Considerations

ΔV

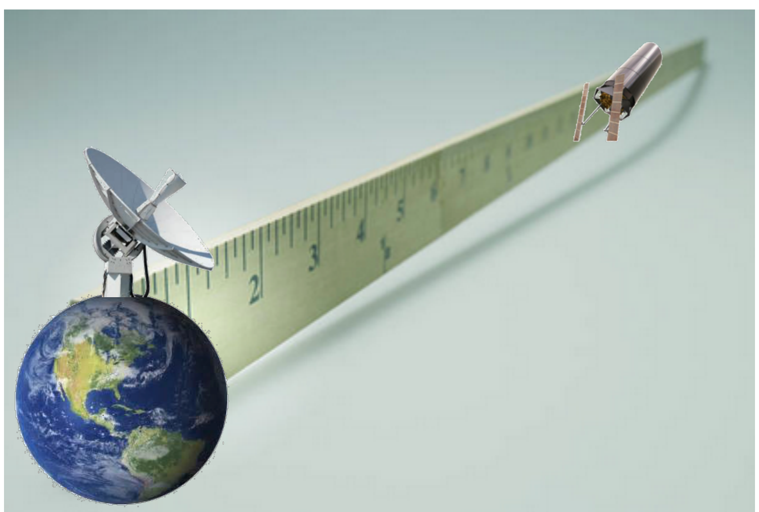
Delta-V Requirements

Duration

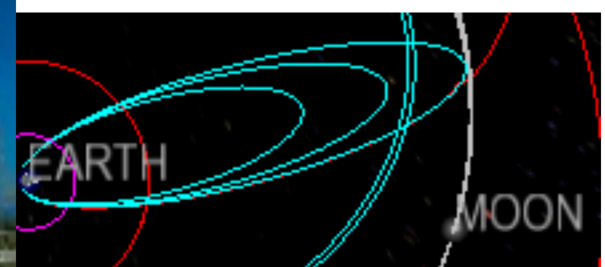


Environments

Thermal considerations



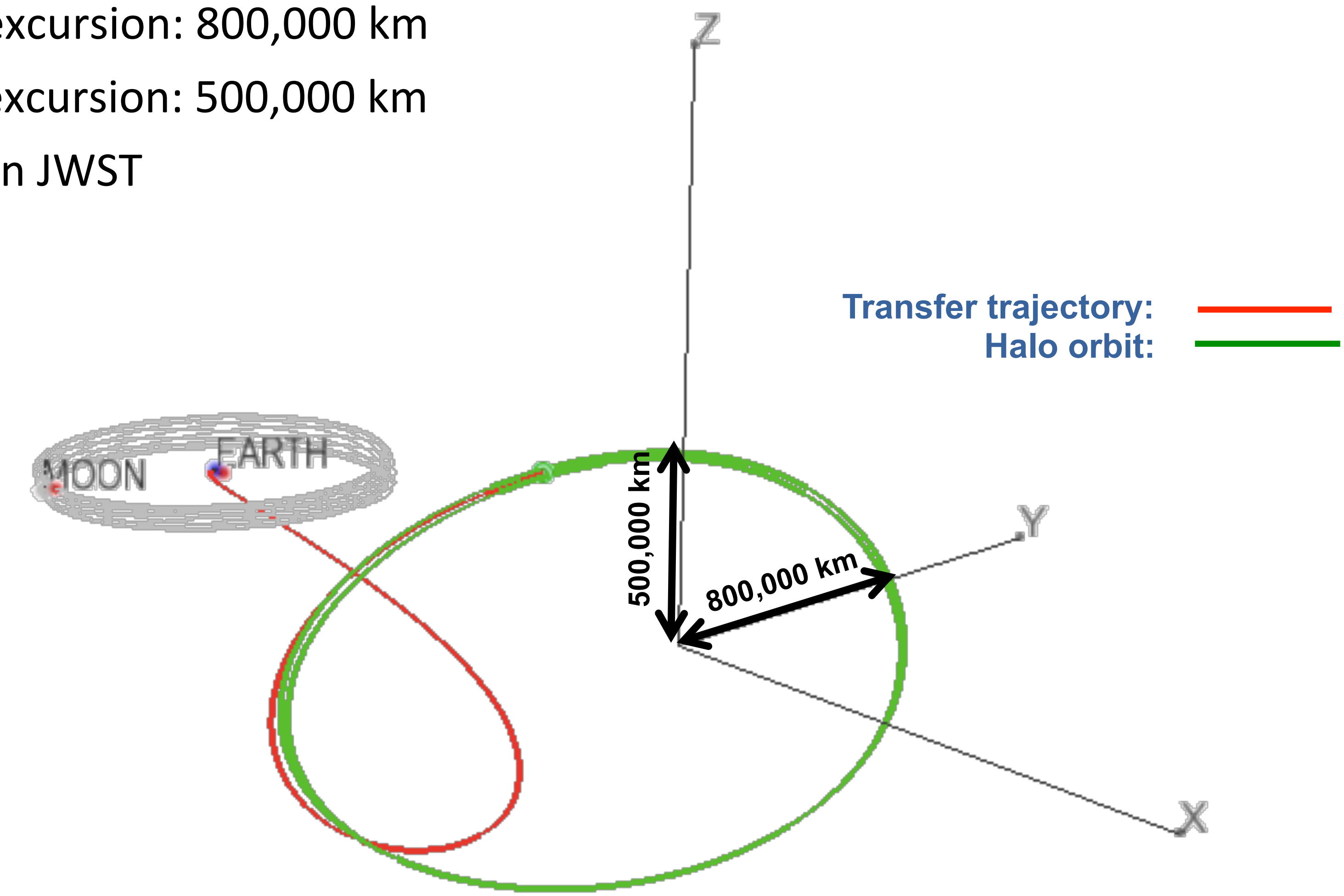
Distance from Earth



Launch vehicle and outbound trajectory

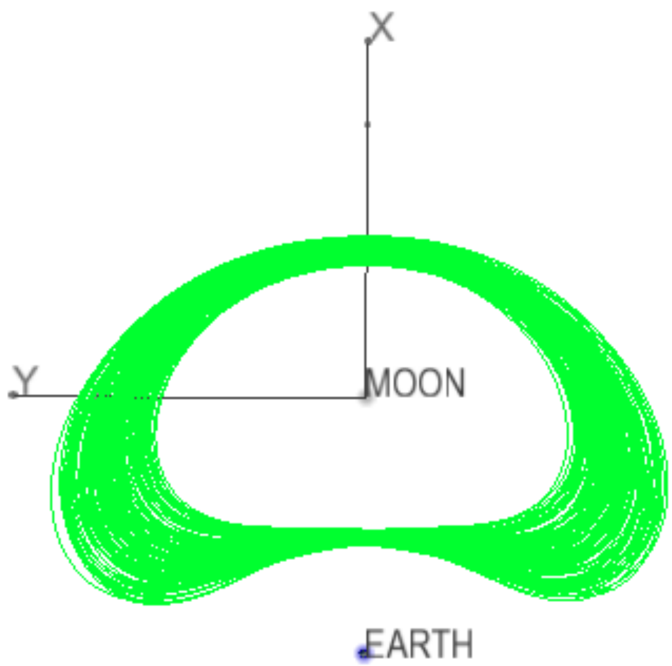
Baseline Orbit: SE-L2

- Sun-Earth L2 Halo
  - Direct orbit (no lunar gravity assist), 0 insertion
  - Max Y-excursion: 800,000 km
  - Max Z-excursion: 500,000 km
  - Based on JWST

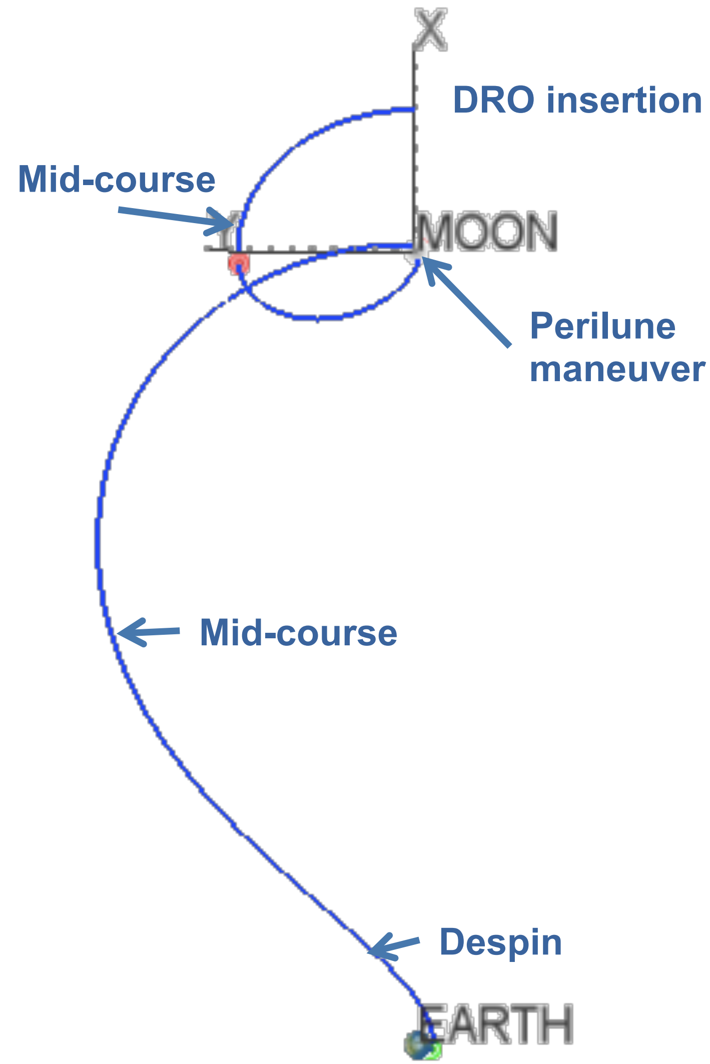


LDRO

- Very stable
  - No disposal required
    - But did include small maneuver in budget
  - Max distance from Earth
    - 500,000 km



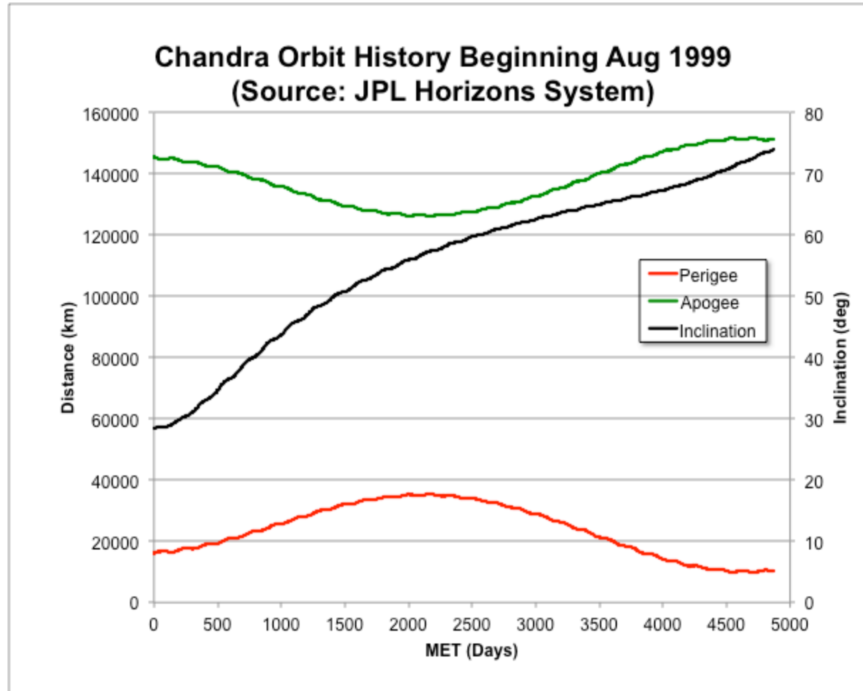
DRO
(Shown in rotating frame)

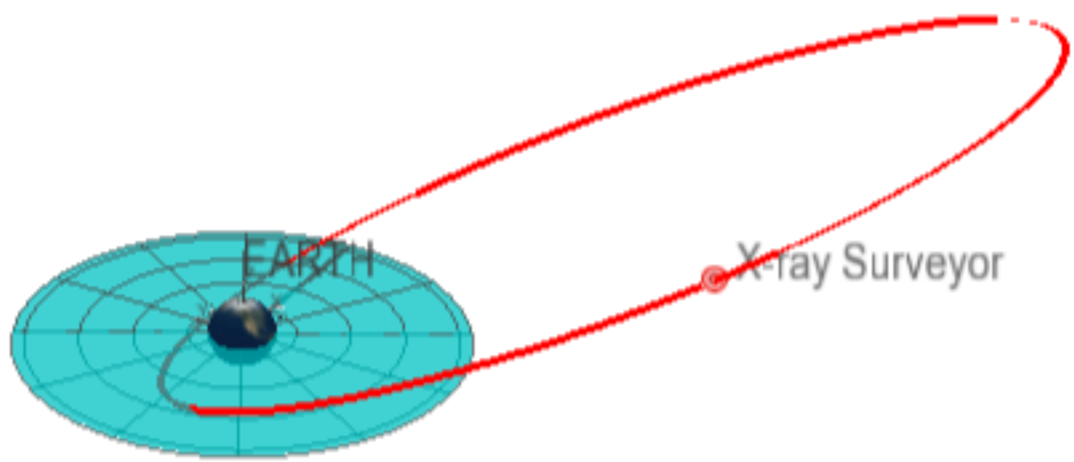


Transfer to DRO
(Shown in rotating frame, based on previous work by Dan Thomas/ED04)

Chandra-Type Orbit (CTO)

- Earth-centered, highly eccentric orbit
  - Placed into final orbit by launch vehicle
  - 16,000 x 133,000 km altitude orbit, 28.5 deg (initially)
  - End-of-life disposal may pose a problem
  - Based on Chandra mission





CTO Disposal

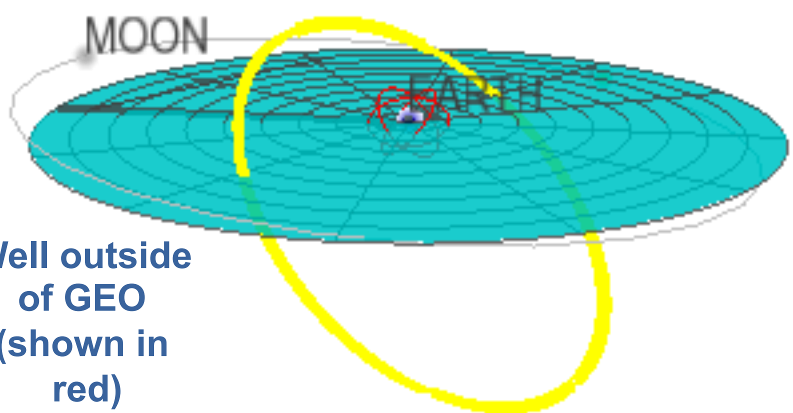
- According to the Orbital Debris Program Office:
  - "The current requirement for the mission you described is to maneuver the spacecraft at the end of mission to a disposal orbit above GEO with a predicted minimum perigee of GEO +200 km (35,986 km) for a period of at least 100 years after disposal."
- 100 years is a LONG time to propagate an orbit, so used Copernicus with Earth J2, moon, and sun as gravitating bodies
    - To be conservative, targeted GEO + 1200 km as minimum altitude
  - This resulted in a target initial perigee for the disposal orbit of about 39622 km altitude (46000 km radius)
- The delta-v for this maneuver is 302 m/s
  - much less than the initial estimate from DAS

We should assume that disposal is required.

Transiting Exoplanet Survey Satellite (TESS)-Type Orbit

- Earth-centered, highly elliptical orbit
  - Approx. 108000 x 376000 km
  - Resonant with the moon
  - Designed for long-term stability
  - Requires lunar gravity assist
  - Maneuver requirements are based on TESS delta-v budget
  - No disposal maneuver required

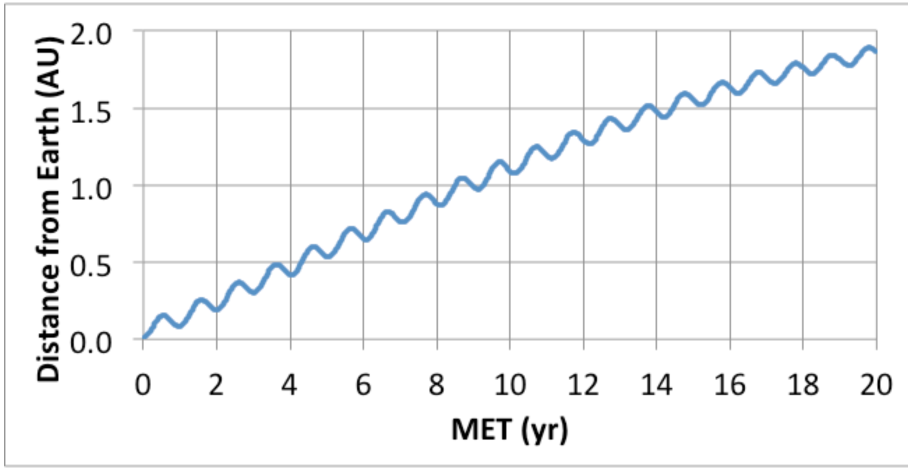
30 years of propagation shows the spacecraft does not pass through GEO (shown in red).

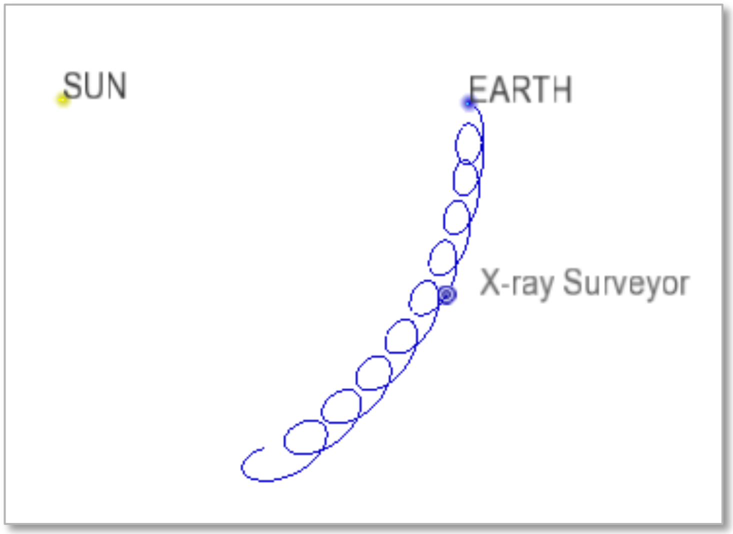


Well outside of GEO (shown in red)

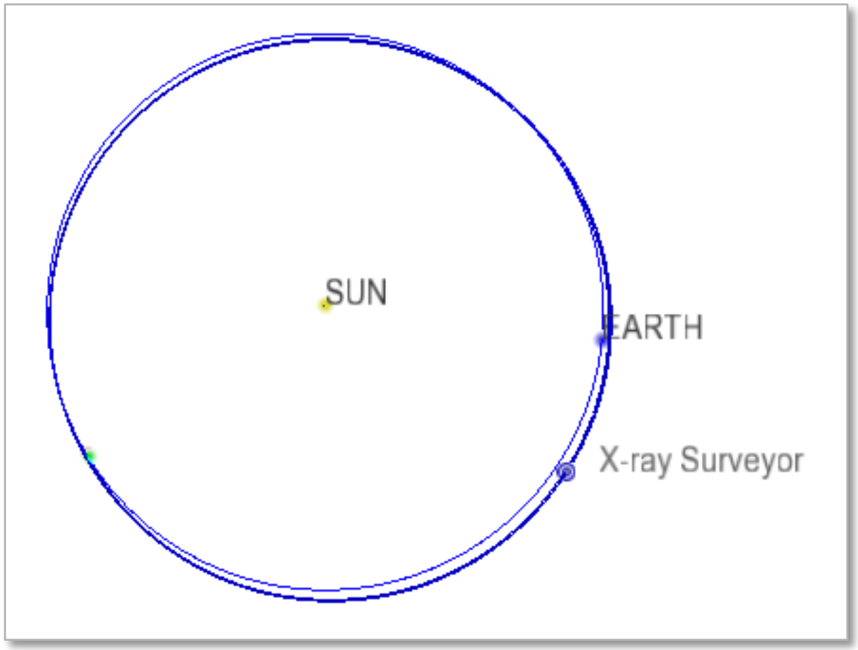
Drift-Away Orbit (DAO), Earth-Trailing

- Launch spacecraft directly into heliocentric orbit
  - No insertion, station-keeping, or disposal maneuvers
  - Distance from Earth to satellite increases over time
  - Based on Kepler mission





Shown in rotating frame



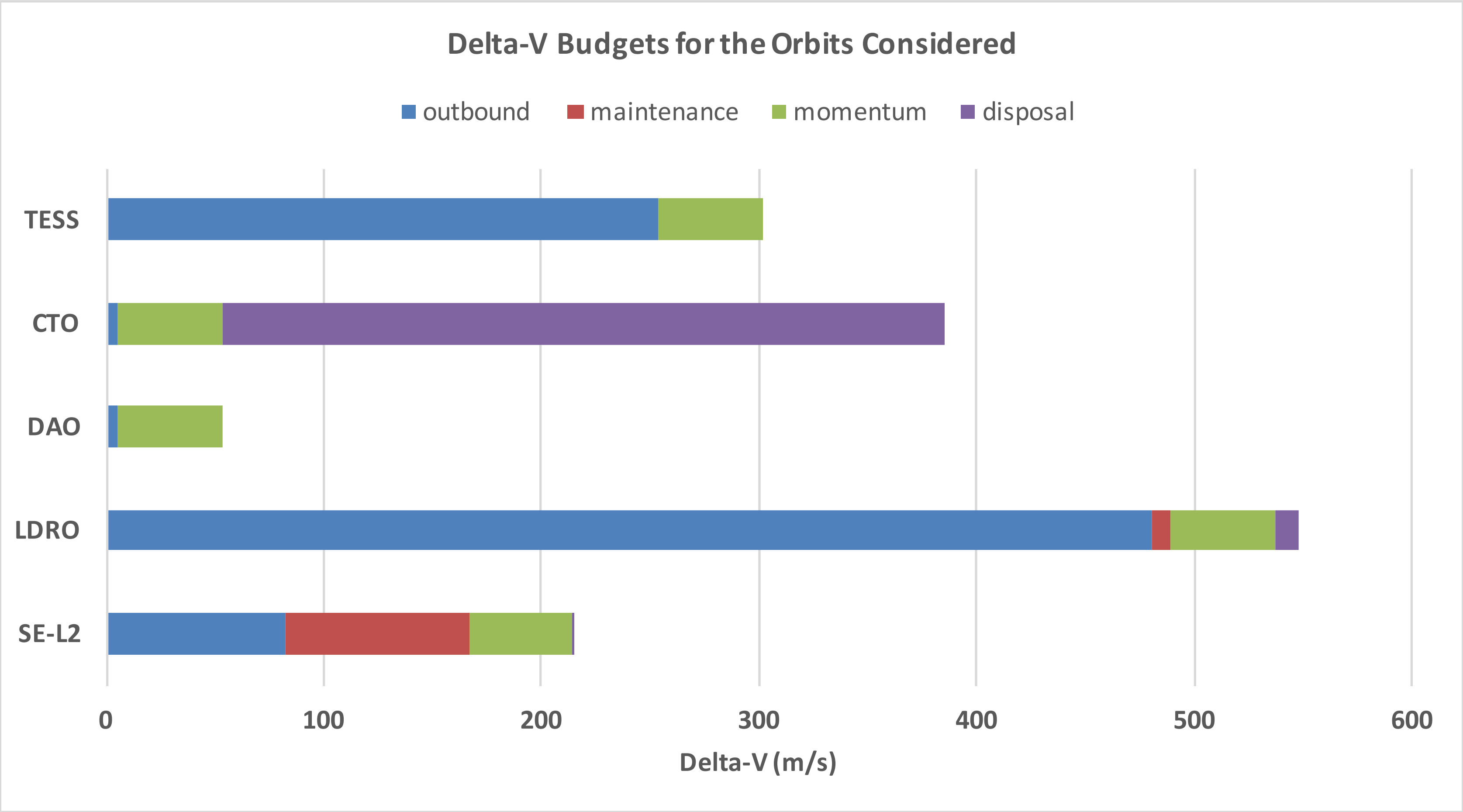
Shown in inertial frame

Orbit Comparison: Notes

- *While the Delta-V budget includes the transfer and operational orbits, the eclipse times are only for the operational orbit*
  - *Momentum unloading Delta-V is a placeholder, with the same value being used for all options (can be revised after orbit downselect)*
- *Assuming all options can fulfill the sky observing requirements*
- *No* consideration given to launch windows

- Results presented below are rough assessments, and for comparing the different options
  - Detailed analysis will be completed after downselect
  - Assessments for the various areas are either supported by analysis (Delta-V, eclipse, duration) or are subjective ratings by subject experts based on experience and basic assessments (thermal, environments, effect of distance on communications)

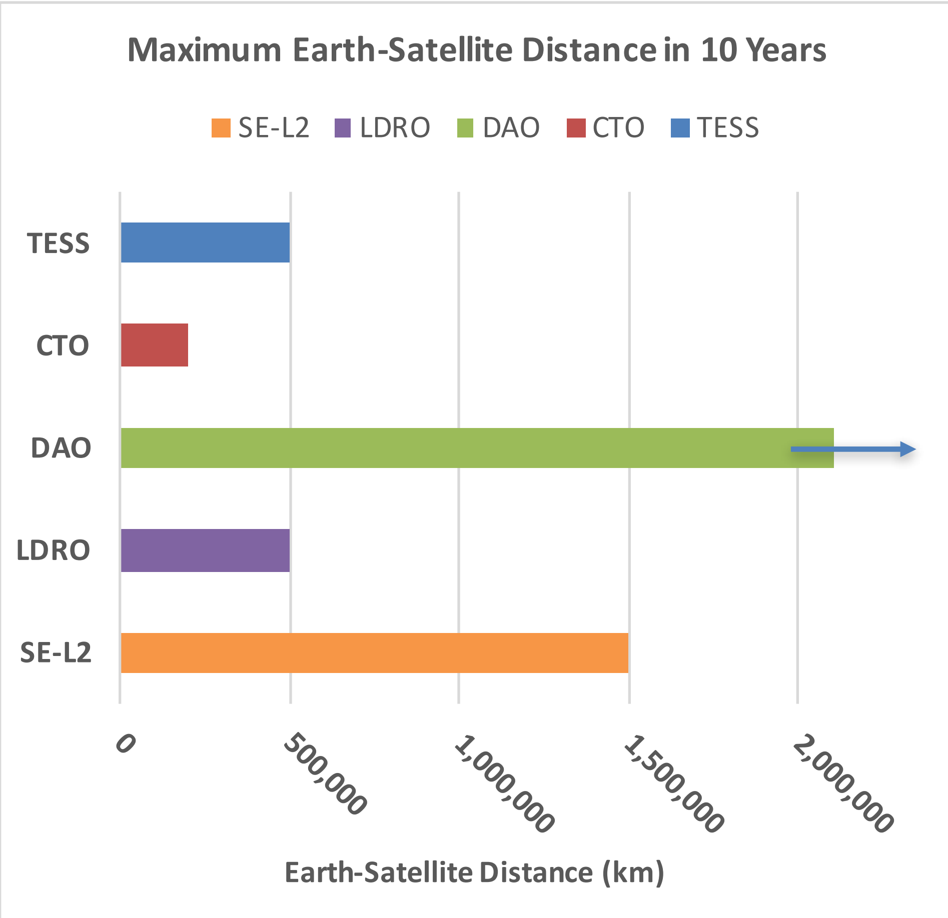
Orbit Comparison: Delta-V

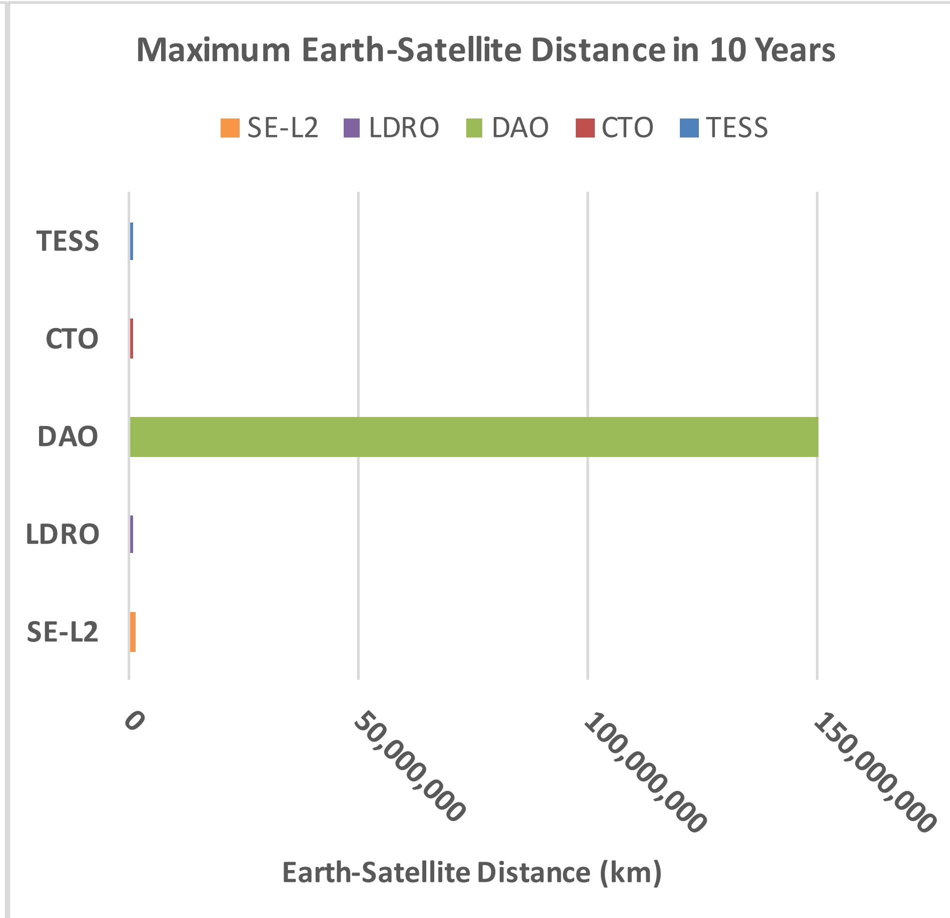


Momentum unloading Delta-V is the same for all options:

- **Reasonable assumption, though CTO probably higher**
- **Revised after downselect**

Orbit Comparison: Distance from Earth

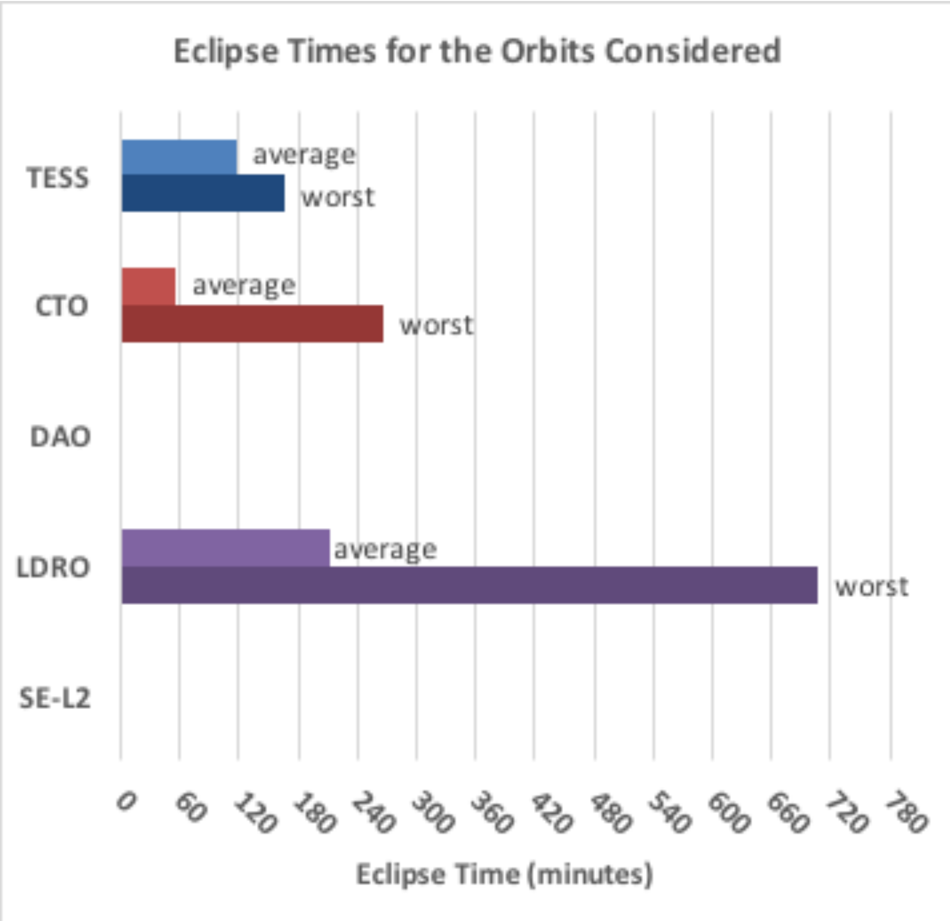


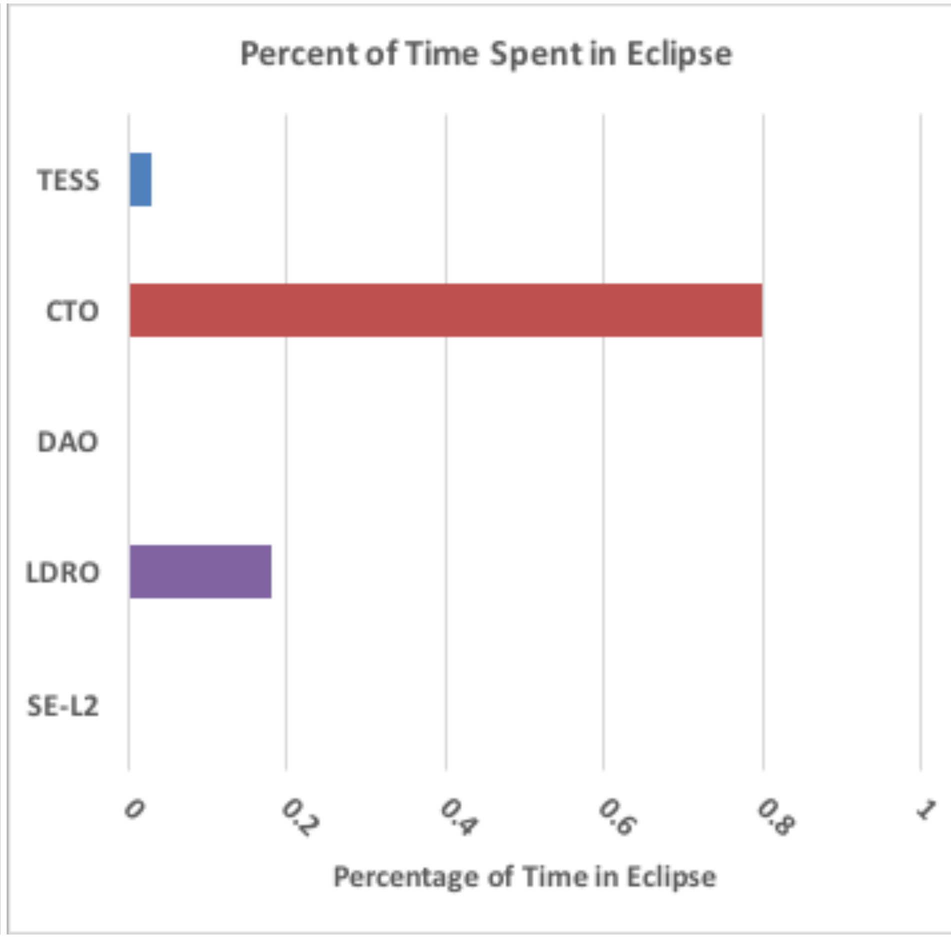


DAO by far the worst -- nearly 1 AU away from Earth after 10 years. Plot on right shows all options to scale.

Note the graph on the left uses an exploded scale. DAO extends to 1 AU!

Orbit Comparison: Eclipse History





Average and worst-case eclipses for the 10-year analysis period.

DAO and SE-L2 have no eclipses, except for possibly during the outbound transfer.

Percent of the time the spacecraft spends in eclipse during the 10-year analysis period.

For comparison, a LEO spacecraft spends about 35% of its time in eclipse.

Orbit Comparison: Figures of Merit (FOMs)

- Subjective ranking of the different options
  - Use the "graduate student" grading scale
    - A = good work
    - B = need to improve
    - C = should you be here?
-

Grade scale	Points
A	1.00
B	0.75
C	0.50

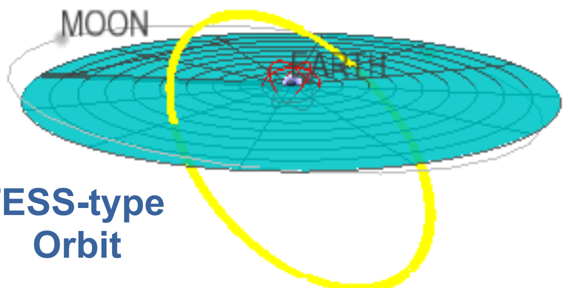
	Total Score	Science	Launch Vehicle	Delta-V	Duration	Thermal	Comm	Environ-ment*
Max Points -->	**100**	20	5	15	15	15	15	15
SE-L2	**93**	A	A	A	A	A	B	B
Drift-away	**81**	A	A	A	C	A	C	B
LDRO	**85**	A	A	C	A	B	A	B
CTO	**75**	B	B	B	A	C	A	C
TESS	**89**	A	A	B	A	B	A	B

*** LEO would get an "A".**

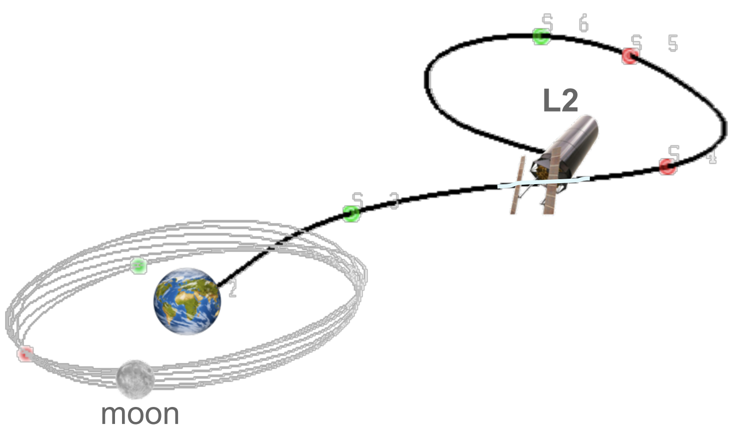
Science inputs pending.
TESS and SE-L2 seem to be the best options.

Orbit Trades: Conclusions

BEST OPTIONS

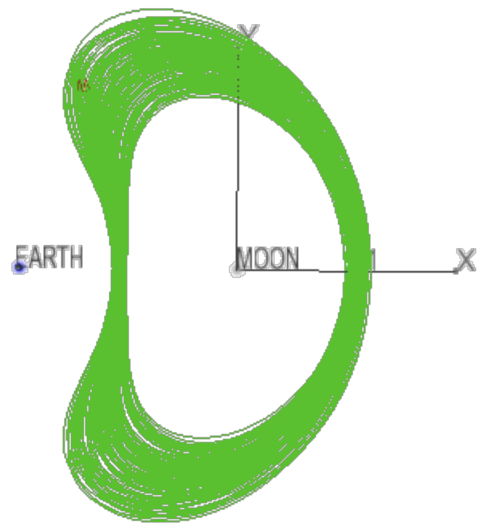


TESS-type Orbit



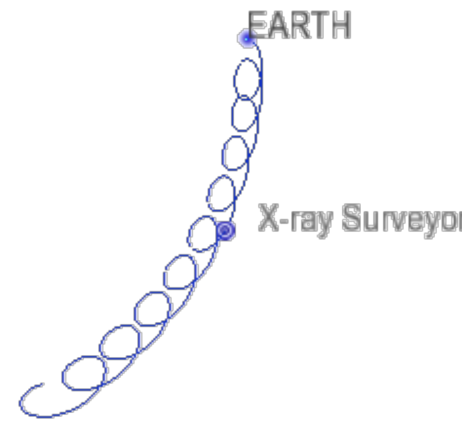
Sun-Earth L2

SATISFACTORY

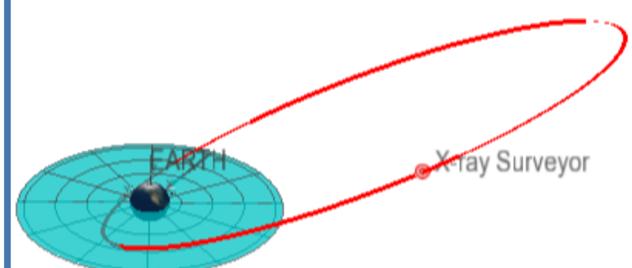


Lunar Distant Retrograde Orbit (LDRO)

CHALLENGING



Drift-Away Orbit (DAO)



Chandra-type orbit (CTO)

Launch Vehicle Selection and Performance

- Contacted NLS
  - Since launch is 2030, actual performance numbers are only useful for getting an idea of the performance available in the future
  - According to NLS, we can be confident that some vehicle will exist that can meet all performance requirements, except for launching such as large payload directly into a TESS-type orbit
    - For TESS, assume only that launch vehicle can place spacecraft onto a TESS transfer, with correct apogee, but with low perigee
      - Spacecraft would have to raise perigee and perform plane change, or use lunar gravity assist (which is the baseline trajectory)

Source -->	NLS quote		NLS website	NLS website	NLS website	NLS website
Orbit type -->	Elliptical Chandra-type		Drift-away	SE-L2 transfer	LDRO transfer	TESS-type transfer*
Altitude or C3 -->	16000 x 133000 km		C3 = 0.61 km2/s2	C3 = -0.7 km2/s2	C3 = -1.8 km2/s2	C3 = -2.05 km2/s2
Burn profile -->	2-burn	3-burn				(r = 6578 x 376300 km)
Atlas V 521	3355	3305	4115	4250	4345	4365
Atlas V 531	3995	3950	4885	5005	5110	5135
Atlas V 551	TBD	TBD	6040	6185	6310	6340
Falcon 9 (v1.1)	TBD	TBD	TBD	3715	TBD	TBD
Delta IV Heavy	TBD	TBD	10490	10735	10945	10585
Falcon Heavy	TBD	TBD	TBD	TBD	TBD	TBD

*** Note: performance data for the Full Thrust option of the Falcon 9 was not available, but is not expected to increase performance.**

ADVANCED CONCEPTS

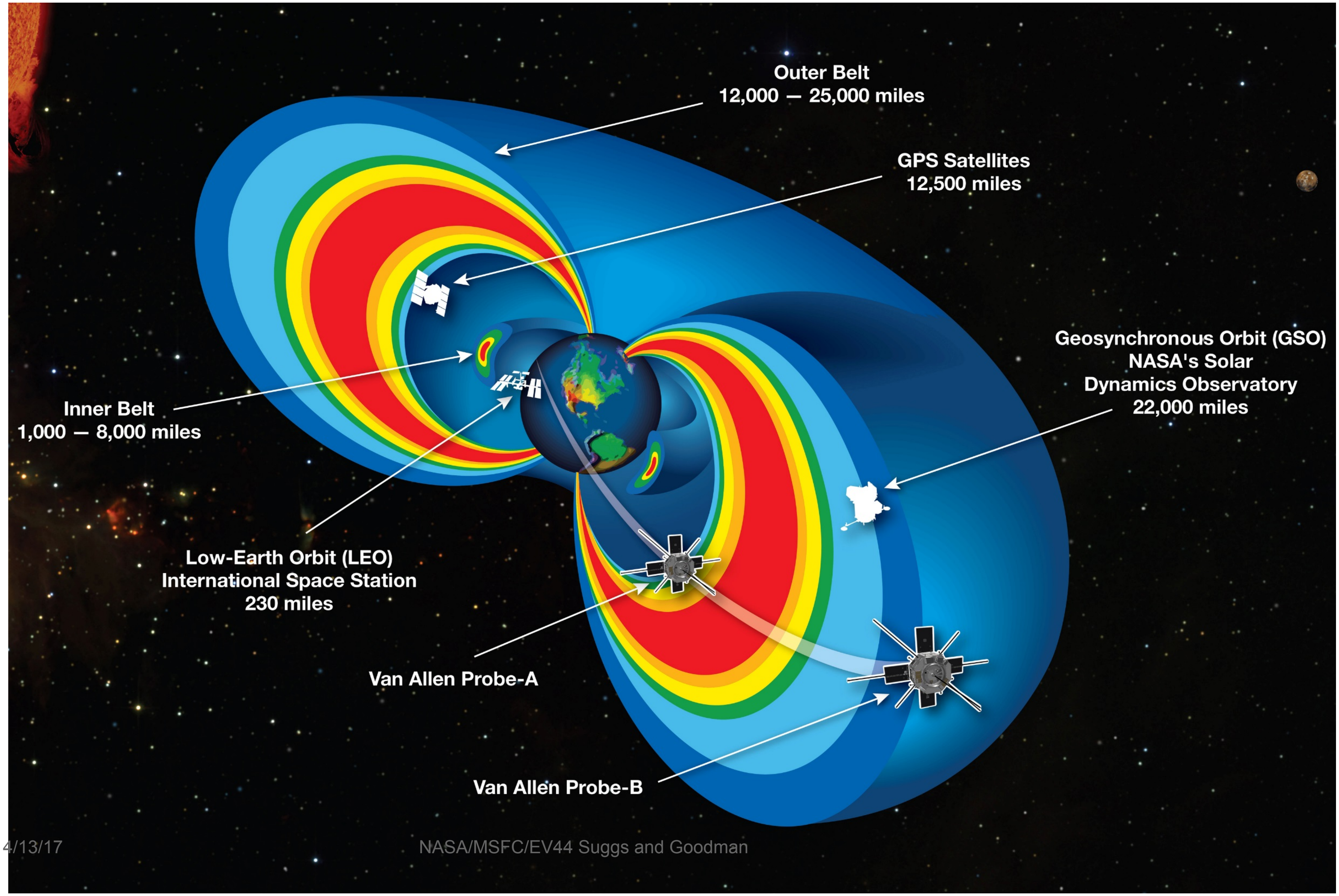
Comparison of Trapped Radiation and Solar Particle Event Environments for Lynx Spiral-out Trajectories

Dr. Rob Suggs, Space Environments Team Lead
Dr. Michael Goodman, Jacobs/ESSSA
NASA/MSFC/EV44
4 April 2017

ADVANCED CONCEPTS

The Problem

- Lynx is considering a spiral-out (multi-burn) trajectory to TESS-like science orbit which causes multiple passes through the Earth's radiation belts.
  - How does the additional radiation dose for these orbits compare with the dose from a single large solar particle event (SPE) when exposed outside of the geomagnetic field?
  - The spacecraft would have to be designed to survive 1 or more SPEs anyway depending on program risk posture.
- Bottom line is an approximately 10x increase in total integrated dose for the thinnest shielding thickness.
  - Trapped belts (mostly electrons) dominate below 0.8 mm.
  - SPE dominates above about 0.8 mm Aluminum shielding.

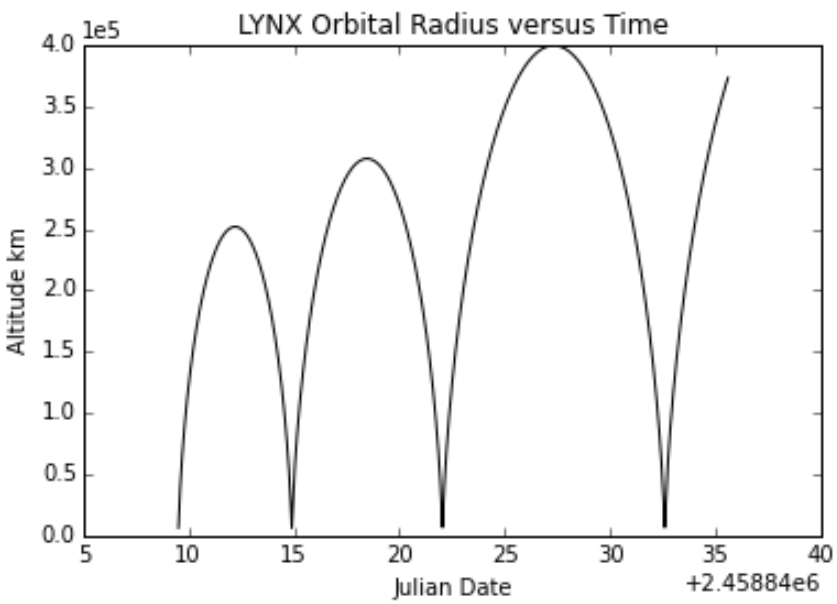

Outer Belt
12,000 — 25,000 miles
GPS Satellites
12,500 miles
Geosynchronous Orbit (GSO)
NASA's Solar
Dynamics Observatory
22,000 miles
Inner Belt
1,000 — 8,000 miles
Low-Earth Orbit (LEO)
International Space Station
230 miles
Van Allen Probe-A
Van Allen Probe-B
4/13/17
NASA/MSFC/EV44 Suggs and Goodman

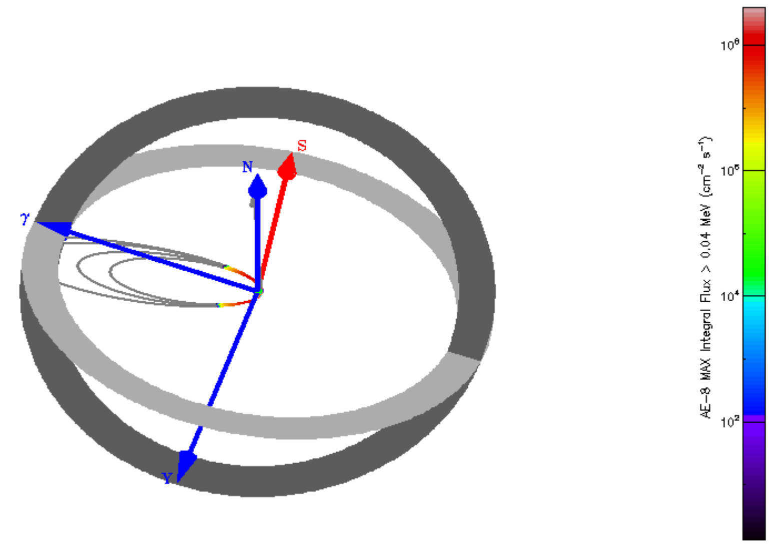
Approach

- Randy Hopkins provided the trajectory as Earth-centered Earth Fixed (ECEF) Cartesian coordinates versus Julian Date.
  - Total mission time is 26.1 days at variable time-step intervals (finer at perigee for better sampling of the radiation belt environment).
  - We requested an inclination of zero to maximize exposure in the radiation belts.
- ESA's SPENVIS web-based space environment tool was used to calculate the trapped radiation environment at each point in the trajectory then calculate the dose versus depth.
  - Used AP8MIN and AE8MAX (worst case solar activity level for trapped protons and electrons, respectively).
  - Used SHIELDOSE 2 module for dose to Silicon versus Aluminum shielding depth.
    - Trapped radiation dose includes protons, electrons, and Bremsstrahlung X-rays generated by electron deceleration in the shielding.
- SPE environment was taken from SLS-SPEC-159 Design Specification for Natural Environments (DSNE) for no geomagnetic shielding.
  - Used ESP/PSYCHIC model with 95% confidence setting and 1 year period.

ADVANCED CONCEPTS

LYNX Mission Profile

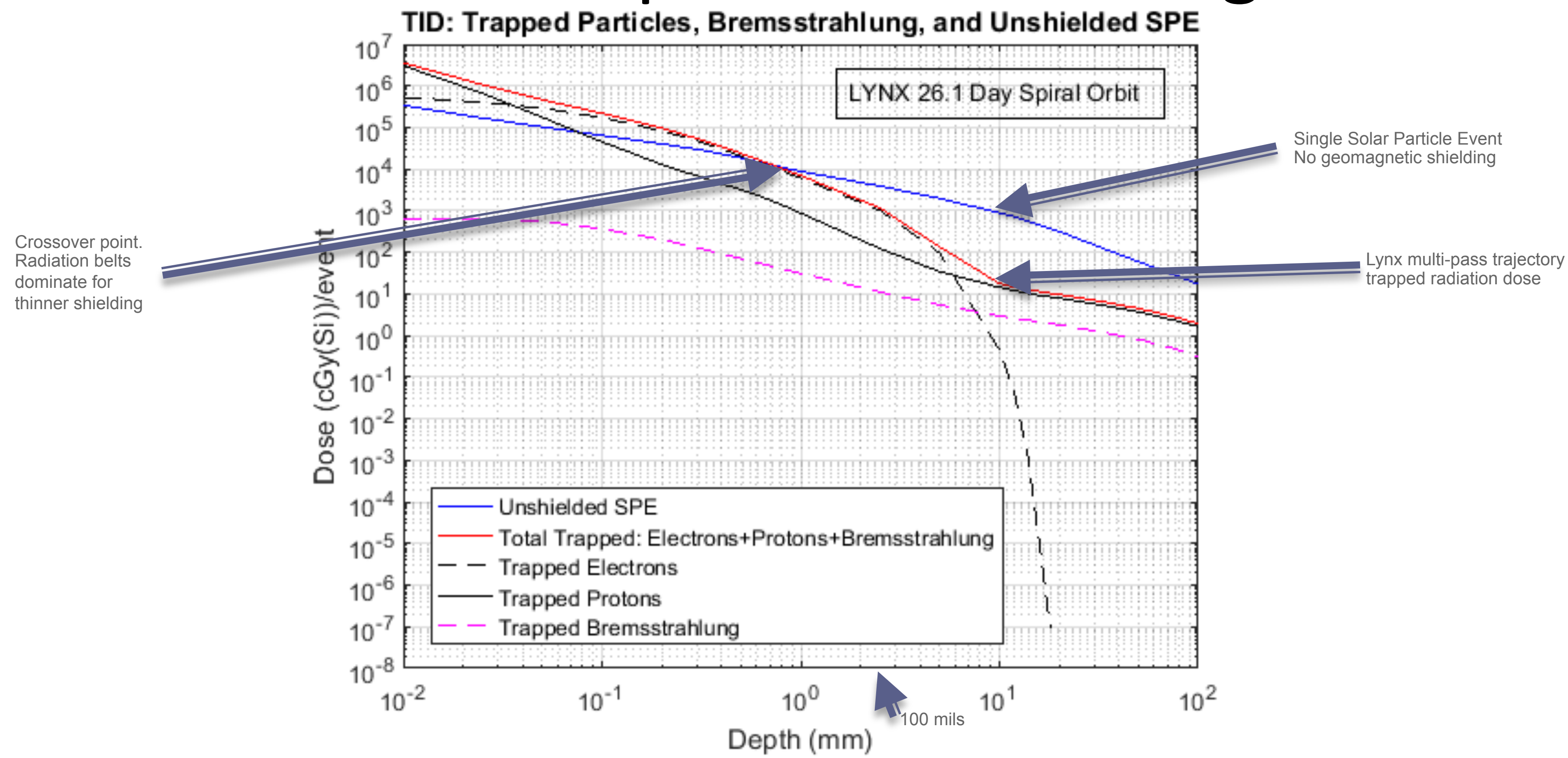




Electron flux

Note that only the lowest portions of the orbits are in the belts

Comparison of total integrated dose (TID) vs depth of Al shielding



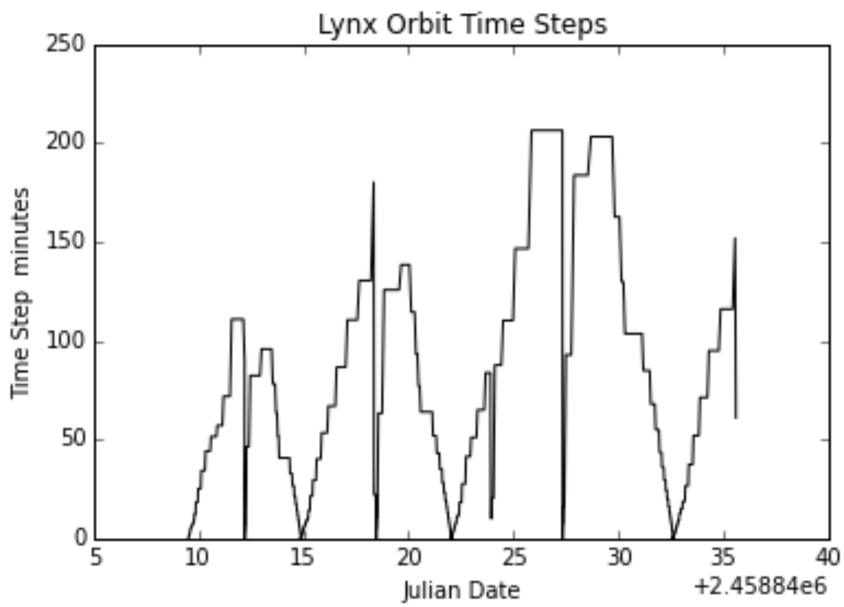
Conclusions

- For total integrated dose (TID) the radiation belt (trapped electrons and protons) dominate for shielding depths below about 0.8 mm of Aluminum.
- For thicker shielding a single large solar particle event (SPE) dominates the total dose.
- We have not compared with a single pass through the belts but, to first order, the dose would be 1/7 of the values reported here (for 7 passes).
- We have not considered dose from galactic cosmic rays (GCR) which are always present and represent lower dose than the SPE.
- We have not addressed single event effects. This is typically dominated by SPE and GCR ions and hardware can be powered down during belt passes to avoid damage.
- We have not specifically addressed solar array damage but will do so in Phase II.

Recommendations/Forward Work

- In Phase II we will evaluate the additional dose to the solar arrays due to the multiple radiation belt passes.

Trajectory Time-Steps



SPENVIS trajectory input file fragment

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*****************************************************
Title: Spiral Out 2
Planet: Earth
Coordinates: GEO
Columns: JDCT, X, Y, Z
Format: CSV
*****************************************************
$$BEGIN
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2458849.50000029,-57.0172,6577.8808,-12.5896
2458849.50000058,-57.2784,6577.8786,-12.5895
2458849.50001893,-73.7480,6577.7262,-12.5836
2458849.50003728,-90.2174,6577.5546,-12.5776
2458849.50005564,-106.6865,6577.3638,-12.5716
2458849.50006842,-118.1580,6577.2195,-12.5674
2458849.50008120,-129.6294,6577.0658,-12.5632
2458849.50009399,-141.1006,6576.9028,-12.5589
2458849.50010677,-152.5715,6576.7304,-12.5547
```

Structures Preliminary Independent Studies

Jay Garcia

Ground Rules and Assumptions

8.0	Property	Value
Structures GR&A	General	Primary structure will be designed to meet minimum strength requirements as stated in NASA-STD-5001B
	Load Cases	Telescope will be designed to withstand Atlas V launch loads (6g axial, 2g lateral)
	Components Analyzed	Structures to be analyzed include: Optical Bench, Spacecraft BUS, Launch Adapter, Translation Table.
	Factor of Safety for Composite Materials	Ultimate Factor of Safety FSu=1.4 (Uniform Areas) FSu=2.0 (Areas with discontinuities)
	Factor of Safety for Metallic Components	FSu=1.4 FSy=1.25
	LYNX Stiffness Requirements	First Constrained Mode > 8Hz Lateral, 15Hz Axial
	Secondary Structures	Assume Optical Bench secondary structures have a mass equal to 20% of the subsystem mass which attaches to the Optical Bench

LYNX Structures (Comments and Questions)

- Optical Bench Fabrication
  - Manufacture using high modulus M55J and T300 carbon composite materials
- Translation table
  - What structural components need to be modeled?
  - Assume metallic fabrication? (Aluminum, Titanium, Steel)
- Mirror and Mirror Mount
  - Recommend composite wrap tubular truss structure
    - Use High modulus M55J and T300 carbon composites
  - Mirror modeled using point masses and Multi Point Constraints
- Magnetic Broom
  - Is this structural?
- CAT Gratings
  - Are CAT Gratings Structural
- LYNX Spacecraft BUS
  - Spacecraft BUS Material Selection
    - Metallic or Composite? (Is thermal CTE important?)
- Sun Shade – Scale from Chandra or Hubble?

LYNX Structures (Comments and Questions)

- LYNX to Launch Vehicle Integration
  - Material Selection for Payload Adapter?
    - Metallic (Truss, Grid Stiffened, Monocoque)
    - Composite (Truss, Honeycomb Sandwich, Monocoque)
    - Forward End Snubbed to Fairing or Un-Snubbed?
  - First constrained mode > 8Hz Lateral, 15Hz Axial (Atlas V Requirements)
  - Launch / Ascent Loads (6g axial, 2g lateral)

Thermal Payload Independent Tasks

Steven Sutherlin

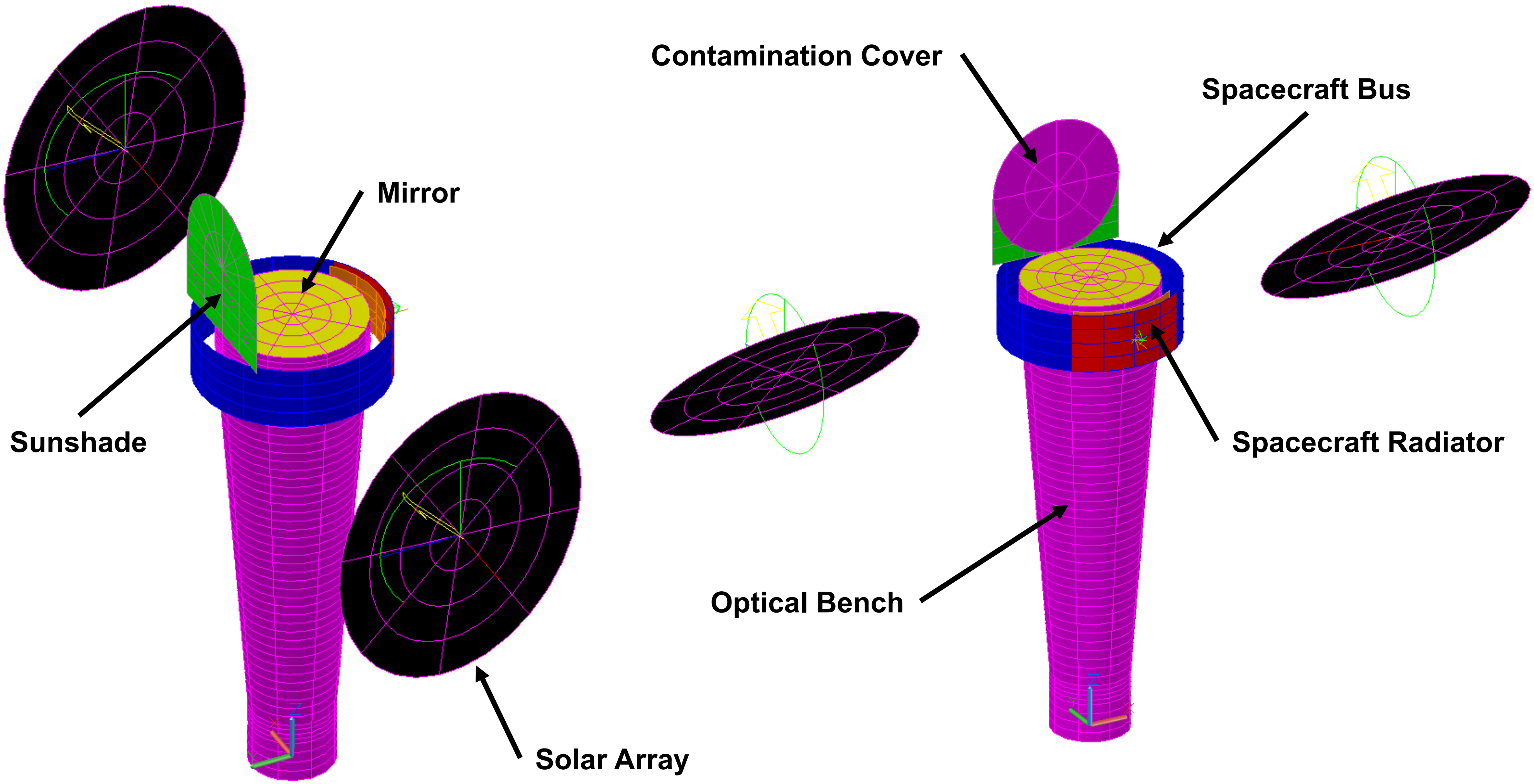
April 4, 2017

National Aeronautics and Space Administration
www.nasa.gov

Ground Rules and Assumptions

Category	Value
Spacecraft thermal control	Thermal control of the spacecraft shall utilize standard, flight-proven features such as MLI, selected surface finishes, foils and tapes; coupled and isolated mounting concepts; optical solar reflectors and radiators; resistance heaters, thermostats and controllers; and pumped fluid loops, cold plates, heat exchangers and fluid radiators.
Instrument enclosure requirements	TBD
Optical bench temperature requirement	300 K
Spacecraft bus temperature requirement	300 K
Mirror heater input power requirement	3000 W (295 K), value provided by customer
Subsystems waste heat load	2000 W
Vehicle orientation	Longitudinal axis not less than 45° from Sun
Recommended thermal environment	Sun/Earth L2
Environmental heat loads	Solar flux at Sun/Earth L2: 1296 W/m^2.
Science payload heat loads	Science payload is thermally isolated from the spacecraft.

Preliminary Thermal Model



Thermal Control Approach

- Integrated spacecraft/telescope thermal model using Thermal Desktop
- Spacecraft bus, optical bench, instrument enclosures
  - Multi-layer insulation with low/medium absorptivity (α) outer layer
  - Resistance heaters, Solar input
  - ~50 W input power to spacecraft bus heaters
  - ~1500 W input power to optical bench heaters
  - TBD input power to instrument enclosure heaters
- Spacecraft radiator surface
  - Subsystems waste heat input to spacecraft bus anti-Sun surface
  - High emissivity (ε) outer surface coating
  - Heat rejection temperature: ~300 K
- Mirror
  - Radiation to space 3000 W (295 K), value provided by customer
- Sunshade
  - Multi-layer insulation with low α outer layer
  - Deployed to fixed position
  - Integral with mirror contamination cover

Spacecraft Sizing Results

Component	Qty	Unit Mass (kg)	Total Mass (kg)	Contingency	Predicted Mass (kg)
Heaters, spacecraft shell	1	10	10	30%	13
MLI, spacecraft shell	1	21	21	30%	27.3
Doublers, spacecraft bus avionics	1	5	5	30%	6.5
MLI, propellant tanks	8	0.25	2	30%	2.6
Total			**38**	**30%**	**49.4**

Future Work

- Develop instrument enclosure thermal configuration (Phase 2)
- Complete the integrated spacecraft/telescope thermal model
- Generate final results
  - Satisfy temperature requirements
  - Estimate total heater power

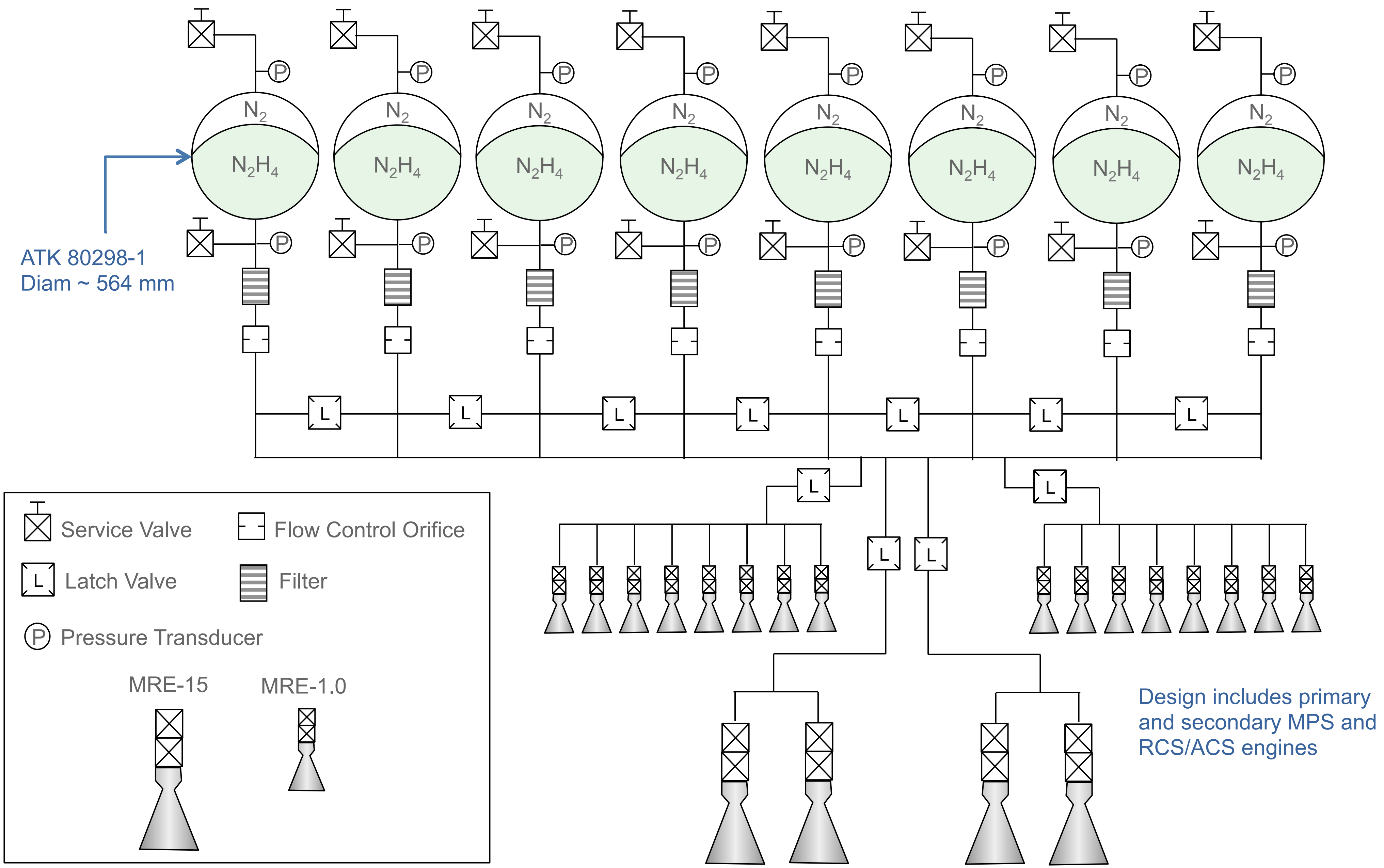
Propulsion Payload Independent Tasks

Tyrone Boswell

Methodology

- Considering high TRL monoprop blowdown system
  - Fuel = Hydrazine
  - Pressurant = Gaseous Nitrogen
- Maneuver Propellant
  - Hydrazine = 494.9 kg (includes 8.75 % extra to fill COTS tank)
- Engines
  - Main Engines: Northrop Grumman MRE-15
    - Thrust = 86 N at 27.6 bar (400 psia), 66 N at 19.0 bar (275 psia)
    - Isp = 228 s at 19.0 bar
  - RCS/ACS Engines: Northrop Grumman MRE-1.0
    - Thrust = 5.0 N at 27.6 bar, 3.4 N at 19.0 bar
    - Isp = 218 s at 19.0 bar
- Mass will be estimated using flight-qualified components
  - Rough estimate made for feed lines and mounts/fittings
  - Reasonable accommodations for possible future servicing will be considered

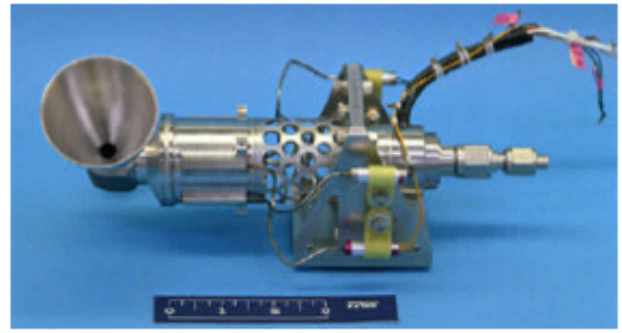
Preliminary Propulsion Schematic



MPS Engine

MRE-15 Monopropellant Thruster

For satellite attitude and velocity control.



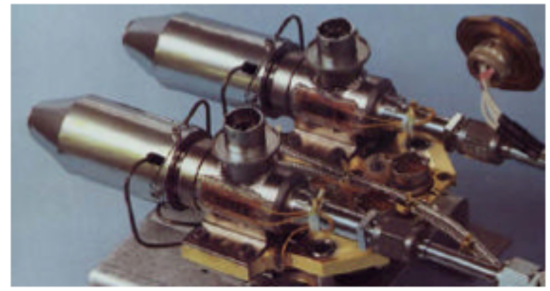
Technical Data

Propellant	Hydrazine
Thrust at maximum operating pressure	86 N at 400 psia
Thrust at 275 psia inlet pressure	66 N
Steady state specific impulse at 275 psia inlet pressure	228 sec
Operating pressure range	138-430 psia
Life (demonstrated)	
• Maximum throughput	970 kg
• Maximum cycles	105,561
Thrust valve power at 28 Vdc	72 W
Weight (STM/DTM)	1.1 kg/-
Envelope (width x length)	119mm x 318 mm

ADVANCED CONCEPTS

RCS/ACS Engine

MRE-1.0 Monopropellant Thruster



For satellite attitude and velocity control.

Technical Data

Propellant	Hydrazine
Thrust at maximum operating pressure	5.0 N at 400 psia
Thrust at 275 psia inlet pressure	3.4 N
Steady state specific impulse at 275 psia inlet pressure	218 sec
Operating pressure range	8-565 psia
Life (demonstrated) • Maximum throughput • Maximum cycles	 544 kg 457,849
Thrust valve power at 28 Vdc	15 W
Weight (STM/DTM)	0.5 kg/1.0 kg
Envelope (width x length)	114mm x 188mm

Spacecraft Programs

DTM - Pioneer, HEAO, TDRSS, FLTSATCOM, EOS, SSTI, SOHO
TOMS, KOMPSAT, ROCSAT, STEP4, STEP1

Propellant Tank

VISIT US ON OUR WEBSITE @WWW.PSI-PCI.COM

ATK

TANK TYPE	MOUNT	LOCATION
Diaphragm	Lugs, 3	Girth

This is a 22-inch spherical pressure vessel constructed of 6Al-4V titanium. Positive fuel expulsion is provided by a reversible ethylene-propylene terpolymer (AF-E-332) rubber diaphragm retained (welded in) at the sphere mid-plane. Mounting is accomplished by three (3) lugs parallel with and adjacent to the sphere mid-plane.

ISO 9001 & AS 9100 REGISTERED

ATK Part Number 80298-1

SIZE: 22.14-inch ID Sphere
SIZE: 562-mm

APPLICABLE DOCUMENTS

Acceptance Test Procedure	50-000255
Qualification Test Procedure	50-000259
Qualification Test Plan	50-000296
Fracture Control plan	54-000049
Qualification Test report	56-000099
Qualification Re-Test Report	56-000106
Cleaning	CPP 3593

DIAPHRAGM INFORMATION

Diaphragm P/N		80-259007-1
Diaphragm Mold P/N		T-1261
Diaphragm Gross Wt		2.47 (1.12 Kg)
Diaphragm Matl Type		AF-E-332
Diaphragm, Material	Note 2	90-000075
Diaphragm Processing	Note 2	90-000087
N-Ray Procedure		1002

Notes:

1: Majority of tooling belongs to ATK
2: Proprietary Document
3: Ea lug ring makes 16 pcs of -7 or 8 pcs of -5
4: Ea tank has one -7 and two -5 lugs
5: Centaur, Titan Centaur
6: Lockheed Matin Astronautics Space Systems
(Formerly General Dynamics Space Systems)
7: Fracture Critical

TANK CHARACTERISTICS

Operating Pressure, psig	485	Total Volume, ci	5,555
Proof Pressure, psig	805	Prop Volume, ci	4,754
Cryo Proof, psig	NA	Max Design Wt, lbs	21.0
Burst Pressure, psig	950	Minimum Wall, inch	0.033

TANK CHARACTERISTICS (Metrics)

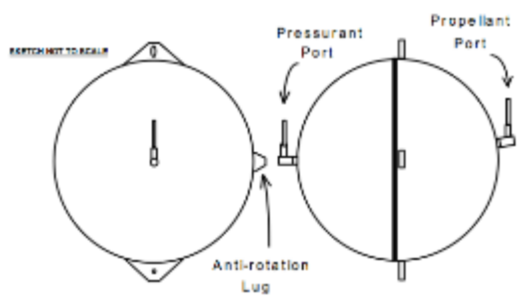
Operating Pressure, Bar	33.44	Total Volume, l	91.03
Proof Pressure, Bar	55.50	Prop Volume, l	77.91
Cryo Proof, Bar	NA	Max Design Wt, Kg	9.53
Burst Pressure, Bar	65.50	Minimum Wall, MM	0.838

FORGINGS

FORGINGS P/N	SUPPLIER	Die No
80-203061-1 (2)		
RING FORGING		**RING SIZE, (Rough Machined)**
80-203009-11, Retainer		24.4 +.06 OD x 21.0 -.06 ID x 1.28 +.06 Lg
80-298063-1, Lug		25.81 +.09 OD x 22.38 -.09 ID x .71 +.06 Lg

TUBE TYPE AND SIZE

TRANSITION	SIZE
80-298001-1, Inlet	.375 OD x .020 Wall
80-298001-3, Outlet	.375 OD x .020 Wall
	9.525 x .5 mm



ACCEPTANCE TESTS

Examination of Product
Pre-Proof Volume Determination
Proof Pressure Test
Post-Proof Volume Determination
Water Expulsion Test
Internal (Diaphragm) Leakag
External Leakage Test
Final Examination of Product
Weight Determination
Cleanliness Verfication
Post Test Inspection

QUALIFICATION TESTS

Acceptance Test
Acceleration Test
Internal Leakage Test
External Leakage Test
Vibration Test
Internal Leakage Test
External Leakage Test
Diaphragm Integrity Test
Internal Leakage Test
Pressure Life Cycle Test
External Leakage Test
Water Expulsion Test
Internal Leakage Test
External Leakage Test
Burst Pressure Test
Post-Test Disassembly & Examination Test

GNC Payload Independent Trades
Rapid Response Considerations

Alexandra Dominguez (EV41)

Additional data used from Dr. Bob Kinsey, ASC (2015 Study)

04/07/2017

Ground Rules and Assumptions

- Phase I focuses on "none-payload specific" trade studies, however a rough idea of spacecraft mass properties/geometry is required to determine bounding disturbance torques and rapid response capability for appropriately-sized actuators. For this reason, *the 2015 study configuration is assumed.*

- The 2015 study target slew rate of 90°/30 minutes and a 100,000s desired continuous observation time are used for bounding calculations.

- Several candidate orbits are considered:
  - SE-L2, Chandra Type Orbit (CTO), LDRO, Drift Away, and TESS

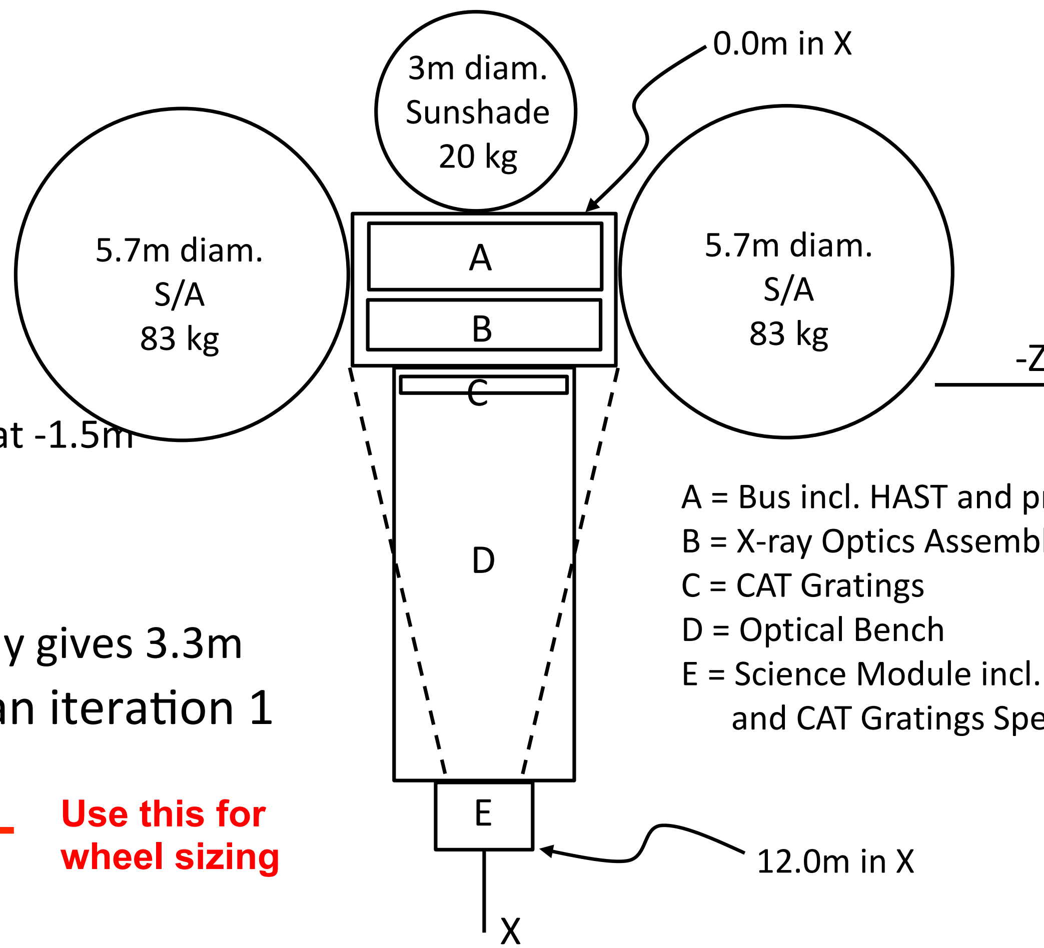
- A 2035 launch date is assumed, however only currently available hardware is considered in this study phase.

- A 10 year mission lifetime is assumed.

Mass Properties Estimate

- Inertias for Y, Z axes ($\perp$ to boresight) are key for determining wheel capability needed to support slew through 90 degrees in 30 minutes.
- Assumptions
  - Solid circular cylinders
    - A+B+C 4m diam. x 2.85m 4572 kg; CM at 1.43m in X
    - D 2.5m diam. x 8.15m 833 kg; CM at 6.93m
    - E 1 x 2 x 2m; 633 kg; CM at 11.5m
    - S/A CMs at 1m; Sunshade at -1.5m
- Total mass 6224kg
  - CM at 3.2m in X
    - Treating A,B,C separately gives 3.3m
  - Inertias a little larger than iteration 1
    - I_{XX} = 14,233 kg-m^2
    - I_{YY} = 87,961 kg-m^2
    - I_{ZZ} = 83,945 kg-m^2

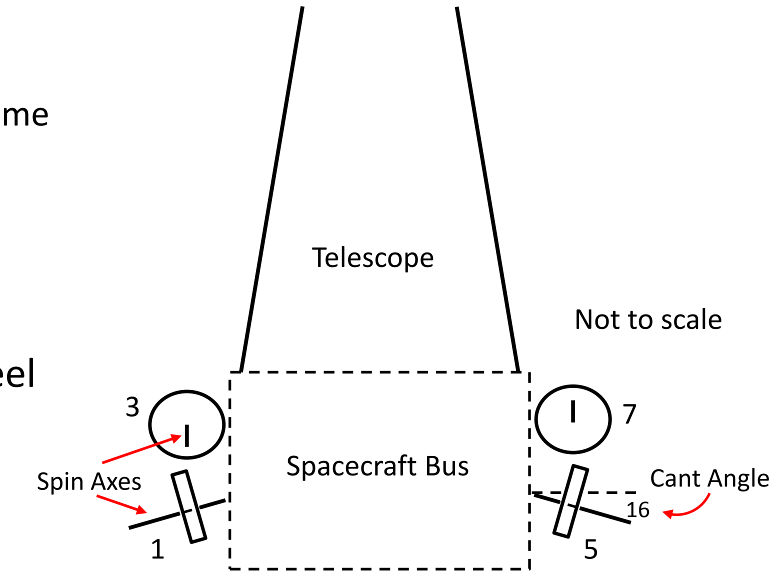
Use this for wheel sizing



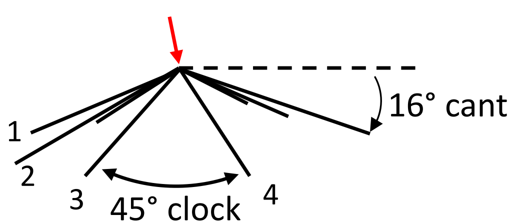
A = Bus incl. HAST and propellant
B = X-ray Optics Assembly
C = CAT Gratings
D = Optical Bench
E = Science Module incl. XMIS, HDXI, and CAT Gratings Spectrometer

Actuator Configuration, Fault Tolerance, and Isolation

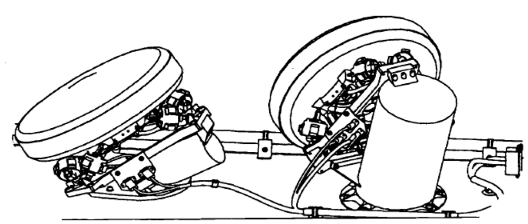
- Wheel Pyramid
  - 8 wheels in “pyramid” configuration; 6 of 8 in operation at a time
    - Cant angle and pyramid orientation can be optimized for more or less capability in any given axis
  - Pairs of opposite wheels shown to the right
  - Spin axis cant angle ~16 degrees for each wheel
  - Spin axis clock angle of 45 degrees between adjacent wheel
  - One isolator per wheel; < 2 kg per isolator.

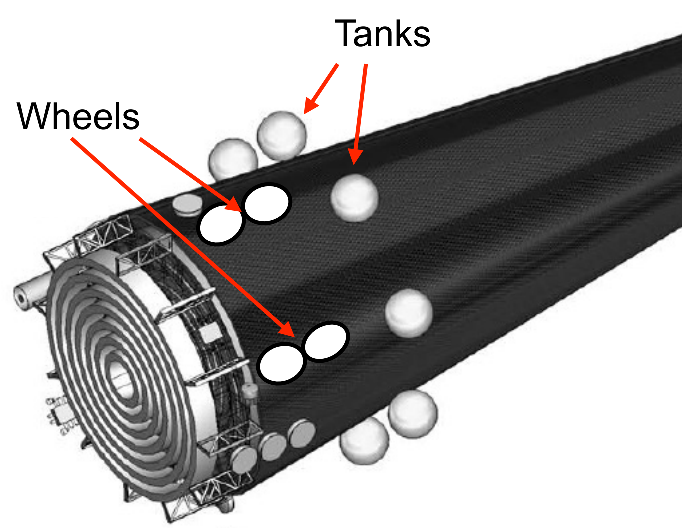


Opposite Pairs of Wheels

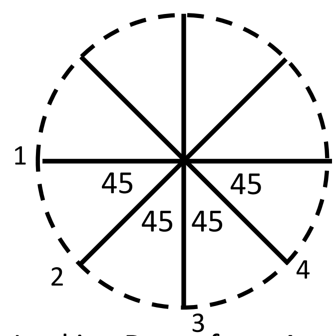


3D View of Spin Axes





- Locations on the vehicle
  - Similar concept used for Chandra
  - Wheel pair at each of four locations
    - 90 degrees around barrel between pairs
  - Isolators mounted to standoffs that provide cant and clock angles.



Looking Down from Apex

Slew Time

- Slew time for worst axis using 4 wheels after a wheel failure.
  - While operating 6 of 8 wheels, only 4 contribute for the worst axis.
- Slew profile used for analysis: max torque to reach max wheel momentum, coast at max rate, then max torque to return to near zero wheel momentum.

Actuator	4-Wheel Max Slew Momemtum (Nms)	4-Wheel Max Torque (Nm)	Min Time to Slew 90 deg (min)	Slew Time with 30% Contingency (min)	Max Momenutm for 90 deg Slew (Nms)			Momentum Margin (%)			Torque for Max Momentum (Nm)			Torque Margin (%)		
					30 min	35 min	40 min	30 min	35 min	40 min	30 min	35 min	40 min	30 min	35 min	40 min
Rockwell Collins TELDIX RDR 57-0	155	0.24	25.62	33.3	153.52	131.59	115.14	1.0	17.8	34.6	0.17	0.13	0.10	40.7	91.5	150.1
Rockwell Collins TELDIX RDR 68-3	184.9	0.2	27.86	36.2	153.52	131.59	115.14	20.4	40.5	60.6	0.17	0.13	0.10	17.2	59.6	108.4
Rockwell Collins TELDIX MW I 100-100/100	272	0.27	25.26	32.8	153.52	131.59	115.14	77.2	106.7	136.2	0.17	0.13	0.10	58.3	115.4	181.4
Honeywell HR-14-75	204	1.09	14.41	18.7	153.52	131.59	115.14	32.9	55.0	77.2	0.17	0.13	0.10	539.0	769.8	1036.0
Honeywell HR-16-75	204	1.09	14.41	18.7	153.52	131.59	115.14	32.9	55.0	77.2	0.17	0.13	0.10	539.0	769.8	1036.0
Bradford Engineering W 45	190.4	0.82	15.96	20.8	153.52	131.59	115.14	24.0	44.7	65.4	0.17	0.13	0.10	380.7	554.3	754.6

A 90 degree slew about the pitch axis in 30 minutes is feasible, however margins are lower than desired given currently available actuators. Good margins (≥100%) can be achieved for a 90 degree slew in 35 minutes with the Rockwell Collins 100 Nms wheel. Alternatively, 100% momentum margin can be achieved using the previously-chosen Rockewell Collins TELDIX RDR-68-3 wheel for a 90 degree slew in 45 minutes.

ADVANCED CONCEPTS

Actuator Specifications

Actuator	Unit Momentum (Nms)	Unit Output Torque (Nm)	Unit Peak Power (W)	Unit Average Power (W)	Dimensions (cm)	Unit Mass (kg)	Missions/Built for Flight
Rockwell Collins TELDIX RDR 57-0	57	0.09	90.00	20	34.5 dia x 11.8 (electronics not included)	7.6 + 1.45 (electronics)	Satellites 1500-5000 kg
Rockwell Collins TELDIX RDR 68-3*	68	0.075	90.00	20	34.5 dia x 11.8 (electronics not included)	7.6 + 1.25 (electronics)	Satellites 1500-5000 kg
Rockwell Collins TELDIX MW I 100-100/100	100	0.1	300.00	35	30.0 dia x 15.9 (with electronics)	16.5	Not Provided
Honeywell HR-14-75	75	0.4	195.00	Not Provided	36.6 dia x 15.9 (with electronics)	10.6	Many
Honeywell HR-16-75	75	0.4	195.00	Not Provided	41.8 dia x 17.8 (with electronics)	10.4	Many
Bradford Engineering W 45	20-70	0.3	64.00	17	36.5 dia x 12.3 (electronics not included)	6.95	Olympus, SOHO, Radarsat, Seastar, Skynet-4, XMM, Integral, Rosetta, ADM-Aeolus

* Selected in original study.

[1] Luke Rinard, Erin Chapman, Andrei Doran, Marc Hayhurst, Michael Hilton, Robert Kinsey, Stephen Ringler, “Reaction Wheel Supplier Survey Aerospace Corporation Report, January 6, 2011.

Disturbance Environment

Torque (Nm)	Candidate Orbit				
	CTO**	SE-L2 Halo	LDRO	Drift Away	TESS*
Solar Pressure	-6.2E-4	-6.2E-4	-6.2E-4	-6.2E-4	-6.2E-4
Gravity-gradient	3.9E-3	n/a	2.3E-6	n/a	2.9E-05
Aero* **	-3.4E-9	n/a	n/a	n/a	n/a
Magnetic	7.1E-7	n/a	n/a	n/a	5.26E-09
TOTAL	**3.3E-3**	**-6.2E-4**	**-6.2E-4**	**-6.2E-4**	**-5.9E-4**

*Gravity gradient, aero, and magnetic torques calculated at perigee (108,426 km)
**Gravity gradient, aero, and magnetic torques calculated at perigee (16,000 km)
***Mean atmospheric density and c_d=2

*Solar Torque Calculation (Solar Constant at 1AU, orientation 45° to Boresight)
(Most stressing case- high CP-CM offset)

	PCM (m)	Area (m^2)	Angle Rel to Sun (deg)	Angle Rel to Sun (rad)	Frontal Area (m^2)	Reflectance	Force (N)	Torque (Nm)
Sunshade	-4.8	7.1	90	1.570796327	7.1	0.7	5.4999E-05	-0.000263995
Solar Arrays	-2.3	51	90	1.570796327	51	0.3	0.000302107	-0.000694846
Spacecraft Bus and Star Tracker	-2.4	8.1	45	0.785398163	5.727564928	0.7	4.43676E-05	-0.000106482
X-ray Optics Assembly	-1	4.725	45	0.785398163	3.341079541	0.7	2.58811E-05	-2.58811E-05
Optical Bench Assembly	3.6	18.75	45	0.785398163	13.25825215	0.7	0.000102703	0.00036973
XMIS, HDXI, and CAT Graing Spectrometer	8.2	2.25	45	0.785398163	1.590990258	0.7	1.23243E-05	0.00010106
Totals		91.925			82.01788687		0.000542382	**-0.000620415**

The spacecraft would experience the largest worst-case environmental disturbance torques in the Chandra Type Orbit (CTO), with gravity gradient torque at perigee having the greatest effect. All other orbits have very similar total disturbance torque magnitudes, with solar pressure torque having the greatest effect.

Momentum Management

- Momentum accumulated in 100,000 s of Continuous Observation Time.
  - Can pause observation for momentum unloading if necessary. Suggested 6 to 8 wheel configuration provides capability to operate for > 100,000 s without unloading.
  - Worst-on-worst analysis assumes worst-case disturbance torque magnitude applied continuously (defines a bounding case)
  - Momentum unload and damping of rates due to orbital insertion/burn maneuvers assumed to be carried out using RCS/ACS thrusters. Estimated required Delta V is accounted for in prop budget, but will need to be refined.

Candidate Orbit	Momentum Due to Disturbances (Nms)	Momentum Due to Slew (Nms)	Total Momentum Accumulation (Nms)	Momentum Margin*
CTO	330	131.6	461.6	-41.1
SE-L2 Halo	62	131.6	193.6	40.5
LDRO	62	131.6	193.6	40.5
Drift Away	62	131.6	193.6	40.5
TESS	59	131.6	190.6	42.7

**4-wheel (worst-case after a failure) max momentum for Rockwell Collins TELXIS MWI 100 = 272 Nms.

Momentum accumulation due to *worst-case* disturbance torques acting over a 100,000 s time period and one 90 degree pitch slew in 35 minutes results in poor momentum margin for the Rockwell Collins TELXIS MWI 100 wheel in CTO, but reasonable margins for other candidate orbits. CTO at apogee looks similar to other candidate orbits.

Recommendations / Future Work

- Recommendation is to carry out a 90 degree slew in 35 minutes with Rockwell Collins TELDIX MWI 100 actuators in an 8-wheel pyramid configuration. This would provide sufficient momentum capacity and torque margin, even in the event of a single wheel failure. Alternatively, could consider relaxing requirement on slew time to ~45 minutes in order to reduce mass of required actuator.
- Chandra Type Orbit has a higher magnitude of disturbance torques at perigee than the other candidate orbits. Excluding this case, all orbit disturbance environments are similar. Actual momentum accumulation due to disturbances is dependent on specific spacecraft attitude in the orbit.
- Refine trade on vehicle rapid response
  - Consider representative observation sequences to better model momentum accumulation
- Update estimates of inertias, geometry, and disturbance environment as the spacecraft configuration is determined
- Develop system model to refine disturbance environment estimate and, later, controller design
- Carry out in-depth dithering analysis

ADVANCED CONCEPTS

Avionics Payload Independent Studies

C&DH, Communications

Pete Capizzo

3-31-17d

National Aeronautics and Space Administration
www.nasa.gov

Avionics

- Ground Rules and Assumptions
  - The spacecraft bus will perform avionics functions including:
    - Guidance, navigation, and control (GN&C), and instrumentation
    - Thermal control and power switching for the science instruments
    - Data storage and data downlink operations
  - The science payload will perform data processing including:
    - Analog-to-digital conversion and data compression and filtering
  - Downlink frequency 1 to 3 times per day
    - Chandra downlink once every 8 hours, for 60 minutes each
    - Assumed about the same link time available for similar mission
  - Total science data collection rate is 240 Gbits/day (2.78Mbps)
  - Total science memory storage desired for 48 hours of data (~500 Gbits)
  - Single fault tolerance for critical systems – mission success
    - Redundancy for most avionics and communications components

Communications Trade

(X, Ka, Laser at CTO and SEL2)

Communications System Parameters:

- Following Chandra's downlink schedule of once every 8 hours, for 1 hour:
  - Data collection of 240 Gbits/day gives 80 Gbits/8 hr to be downlinked.
  - 80 Gbits downlinked in 60 minutes requires a rate of 22.2 Mbps.

- Using DSN 34m dish ground station parameters:
  - 54.0 G/T for X-Band and 65.7 G/T for Ka-Band

- Using the Mercury Messenger like Phase Array antennas for science downlink:
  - with a gain of 24.7 dB for X-band and 26 dB for Ka-band.

- Using LADEE LLCD 100nm Laser Comm system:
  - Assuming about 30 dB margin required with 30 dB atmospheric attenuation.

Conclusions for SEL2:

- X and Ka band PA systems will result in similar system mass, with Ka being slightly better.
  - PA size about 0.25m, 25 and 20 watt RF power required respectively.
- Laser comm system will be significantly lighter:
  - 10 cm aperture, 5 watt RF power
  - But requires much greater pointing accuracy, and a pointing gimbal is required
  - Highly dependent on weather conditions, SEL2 probably to far (per SCAN), uplinks even harder

Communications Trade

Communication System Trade Chart, for downlink rate of 22.2 Mbps

	Chandra Like Orbit	Margin	SEL2	Margin
Range	133,000 km apogee		1.5x10^6 km (0.01 AU)	
X-band Power	1 watt	10 dB	25 watt	3 dB
Ka-band Power	1 watt	12 dB	20 watt	4 dB
Optical Power	0.5 watt	40 dB	5 watt	30 dB

- At SEL2, minimum margin required is 3 dB for X and Ka band, 30 dB for optical assumed.
- In Chandra like orbit, the low power and high margins mean greater link rates can be achieved.
  - Over 100 Mbps at 1 watt RF.

Preliminary Avionics

- **CD&H Approach**
  - Baseline JPL Mars Orbiter Computer
    - Designed for long life in deep space environment
    - Handles similar communications requirements
  - Spacecraft bus includes data storage unit for recording 1Tbits of data
    - Baseline EADS Astrium Coreci mass memory unit
    - Provides 1 Tbits of data storage, at 1.4 Gbps
    - Double the 500 Gbits required for 48 hours
- **Communications Approach**
  - For SE-L2, baseline Mars Pathfinder EDL communications system
    - Includes both X-band and Ka-band systems, with TWTA power for each
    - Replaced the 1M parabolic antenna with 4 Phased Array Antennas
    - No pointing mechanism required (no pointing vibrations)
  - For CTO, baseline Messenger like communication system
    - Uses X-band PHA also, but lower power SSPA
    - Provides hemispherical MGA and LGA systems for telemetry and backup

Preliminary Avionics

- Results
  - SE-L2 Communications mass: 55 kg, 369 W
    - Will be similar mass for Drift-away orbit
  - CTO Communications mass: 28 kg, 128 W
    - Will be similar mass for LDRO and TESS orbits
  - CD&H and Instrumentation mass: 113 kg, 282 W
    - Same for either orbit, includes 2-FC, MMU, 3-DAUs, and 4-ACS controllers
- Future Work
  - Investigate using other avionics components to save mass and power
  - 60 minutes of X-band on DSN needs to be verified
  - Continue to investigate Laser/Optical communications
- Note – Jeffery Hayes discussion on future communications capability:
  - On February 28, Jeffery Hayes of HQ Astrophysics division presented a Space Communications and Navigation (SCAN) briefing, and discussed the use of the DSN.for the Lynx project.
  - Both S-band and X-band bandwidth will become much more restricted in the near future due to commercial use
  - Jeffery suggested we consider using all Ka-band for greater bandwidth and higher data rates available
    - Which we can do, but Ka-band equipment is presently limited, and some estimating will be needed
  - Optical communications capability is being development, but is slow going and may not materialize as a viable option
    - Has been successfully demonstrated from lunar orbit by the Laser Lunar Communications Demo (LLCD) on the LADEE mission at ~620 Mbps
    - Optical links from Lunar distances to ground is reasonable, but probably wont work from further out (SE-L2) distances.

Mechanical Systems Payload Independent Tasks

Justin Rowe

GR&A - Translation Table

Category	Value
Instruments' focal plane location	X-Ray Calorimeter and CAT grating planes will be coplanar
CAT Grating Location	Not required on Translation Table
Horizontal translation accuracy	0.0002"
Vertical Translation distance	0.4"
X-Ray Calorimeter instrumentation locations	All instruments (coolers, power, etc) requiring to be less than 1 meter from Dewar Assembly will reside on the Translation Table
Enclosure	Translation Table, science, and supporting instruments will be fully enclosed
Launch Locks	Used until science is activated

GR&A - Inner Optics Door

Category	Value
Service Life	Single use
Pressure	Pressure in Optics compartment, leakage allowed
Open/Closed position	Opened door must reside within optical bench and outside of optical path
Door position monitoring	Secondary monitoring device will be used (Chandra Heritage)
Material	Composite or Metallic

Will be sized by allowable leakage rates, petal diameter, and launch loads

GR&A - Outer Optics Door

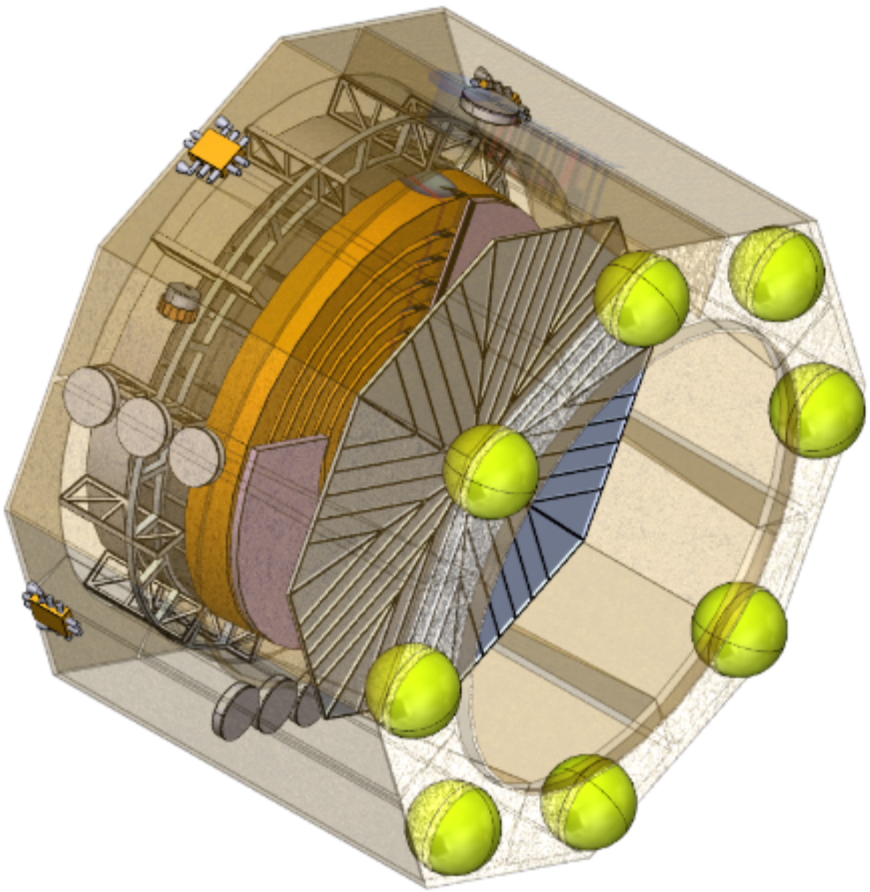
Category	Value
Service Life	Single use
Pressure	Pressure in Optics compartment, leakage allowed
Open/Closed position	Opened door must open beyond optical path and serve as sunshade
Door position monitoring	Secondary monitoring device will be used (Chandra Heritage)
Material	Composite or Metallic
Translation Device	Dual stepper motors (small, reliable, relatively high torque)

GR&A - CAT Gratings

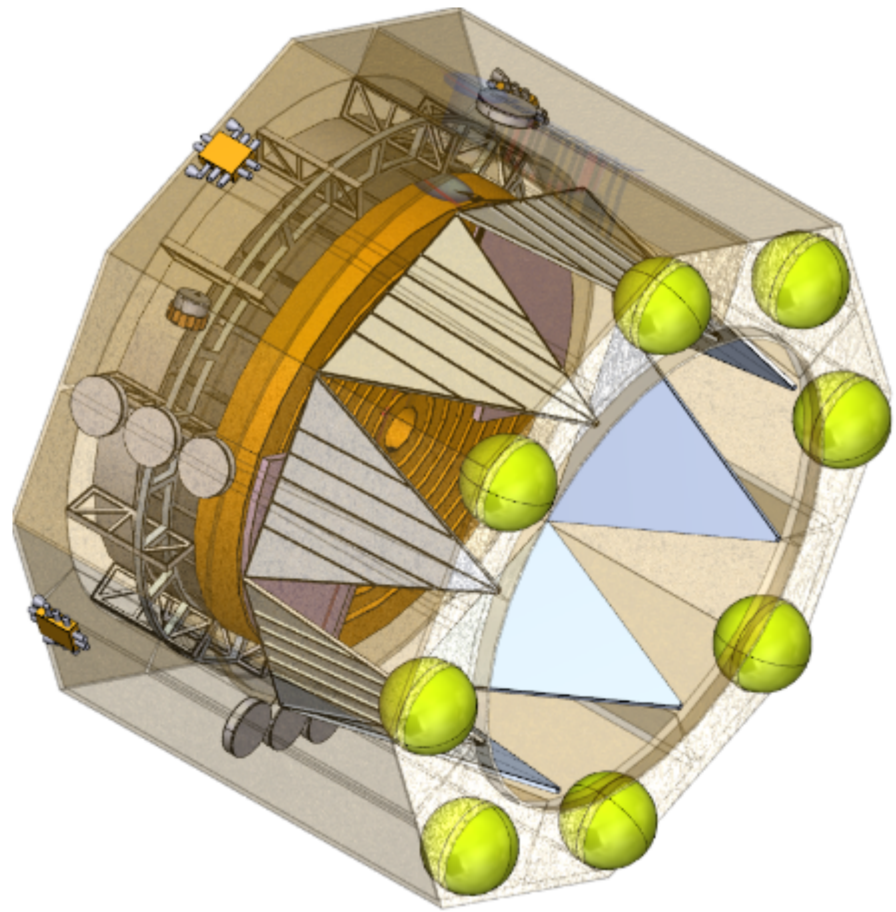
Category	Value
Operation range	Grating must swing into and out of optical path multiple times
Position during launch	Stowed
Accuracy and precision	Large alignment tolerances
Neighboring structure and mechanisms	Inner door will remain outside of operation range
Door position monitoring	Secondary monitoring device will be used (Chandra Heritage)
Grating size	4 Sections covering 3000 cm^2 (about half of optic area)
Translation method	Compact Linear Actuators

Preliminary Mechanical Components

Aft Door – Single Post-Launch Deployment



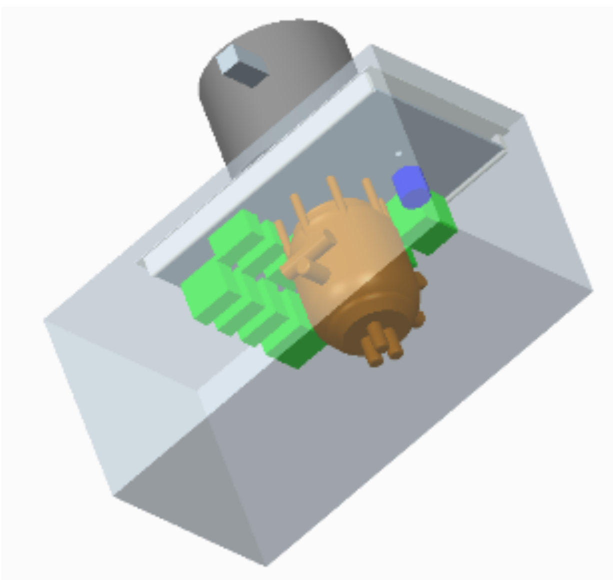
Aft door closed



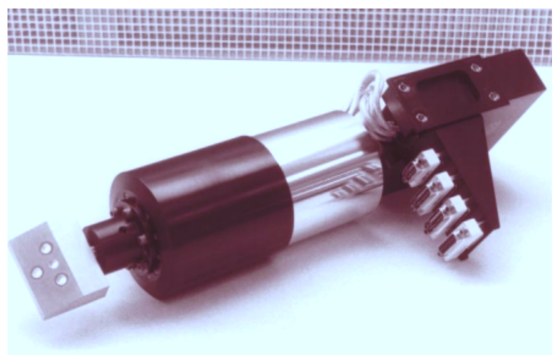
Aft door open

Mechanical Components

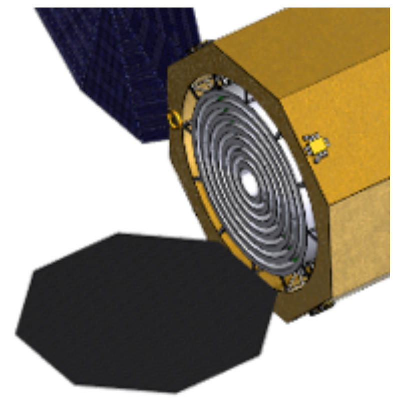
Translation Table
Regular Use



CAT Grating Actuators
Regular Use



Sunshade
Single Deployment



Preliminary Sizing Results

Translation Table	Qty	Unit Mass (kg)	Total Mass (kg)	Contingency	Predicted Mass (kg)
Translation Stage	2	30	60	30%	78
Focusing (vertical travel) Stage	4	4	16	30%	20.8
Secondary Structures and Support	1	25	25	30%	32.5
Enclosure	1	100	100	30%	130
Total			**201**	**30%**	**261.3**

Actuators and Motors	Qty	Unit Mass (kg)	Total Mass (kg)	Contingency	Predicted Mass (kg)
Inner Door Stepper	8	1.5	12	30%	15.6
Outer Door Stepper	2	2.5	5	30%	6.5
CAT Grating Actuator	4	2.5	10	30%	13
Total			**19.5**	**30%**	**35.1**

Questions & Further Research

- **Sizing and fixturing of CAT grating for mass & inertial calculations needed in sizing**
- **Sizing of translation table – notional has it at 750mm but could that be narrowed to 600mm without causing significant issues, to drastically increase commercially available options and decrease mass?**
- **Diameter of the focal regions within the optical bench compared to the inner diameter of the focal bench (ie: how much space is available for the inner door segments once deployed?)**
- **How far and in which directions should the CAT gratings translate within the optical bench?**
- **What changes from Chandra design do we need to implement to help eliminate stray light and high-energy protons? (See Chandra “Lessons Learned”)**

Power Payload Independent Studies

Leo Fabisinski

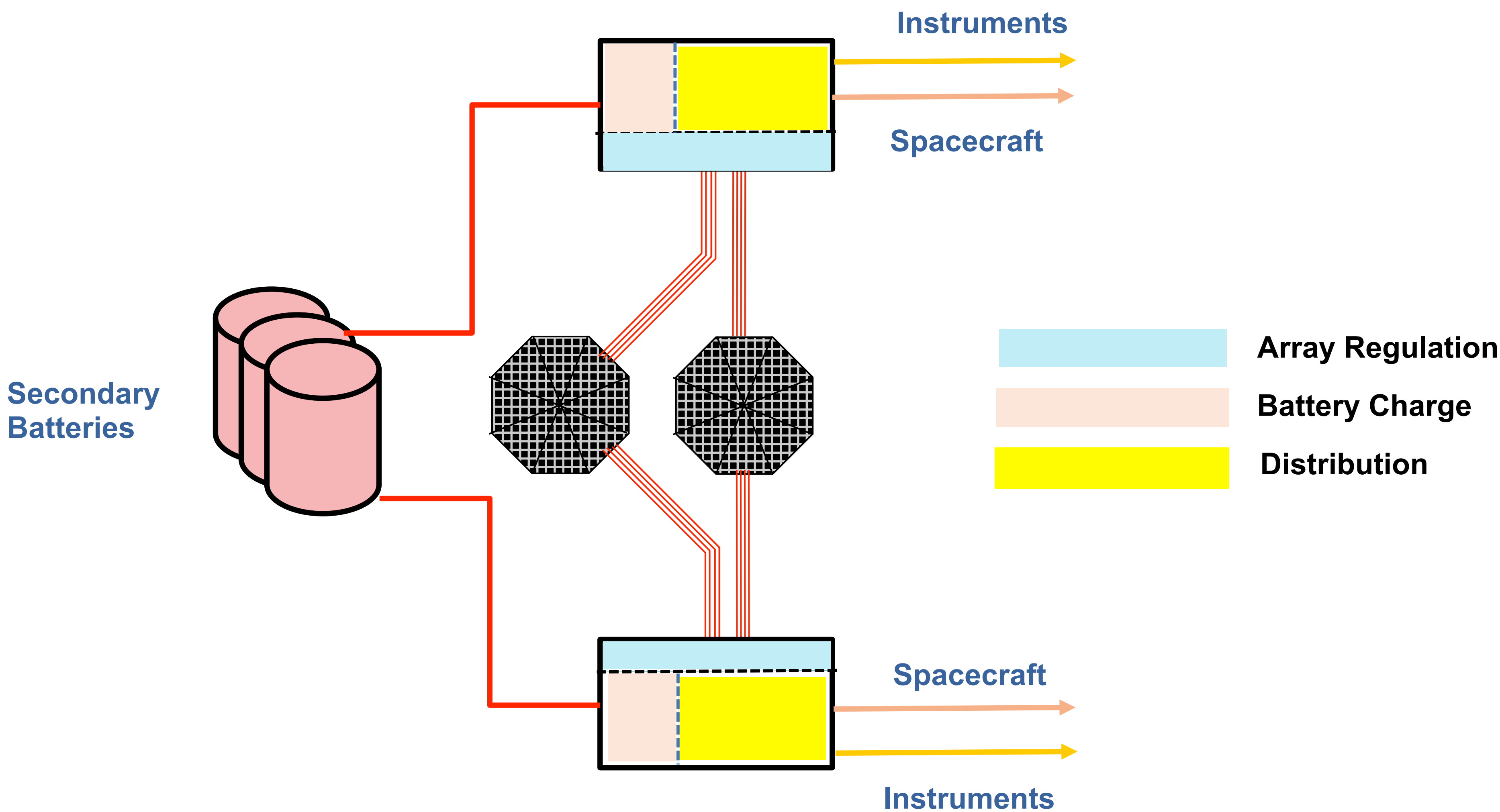
Ground Rules & Assumptions

Ground Rule / Assumption	Value
Power provision	Power System will store, generate, manage/condition and distribute power to all subsystems and payloads on the vehicle
Maximum Battery Power Time	180 minutes
Bus voltage	120V / 28V Nominal.
Power during initial checkout / solar array deployment	Power will be provided to all attached architecture elements during initial checkout (1 hr) and solar array deployment if required. Full power will remain available during final orbit insertion.
Payload circuits	20 switched circuits provided to payload
Overload protection will be provided	for all critical functions (should consider resettable fuses)
Fault tolerance	No single fault will allow the vehicle to enter mission critical failure mode
Ground reference	A common ground reference will be provided across all subsystems
Secondary battery charge/discharge efficiency	95%
Secondary battery max depth of discharge	60%

Approach and Methodology

- Solar arrays are sized for 10 year full-power life.
- UltraFlex array chosen for structural strength and light weight at moderately large power generation levels (App. 10kW).
- Structure is sized for required structural loads. Target orbits requiring impulsive insertion will require more robust (and heavier) structure. Target orbits requiring extended exposure to radiation belts will require greater photo-voltaic area.
- Fully redundant power electronics required for Risk Class A mission.
- Energy storage is sized to provide power for 180 minutes. This provides 3 hrs of battery power for servicing the arrays at a later time.
- Integrated power electronics (solar array regulation, battery charge control and power distribution) are sized using components designed for use in the Orion vehicle.
- Cabling and harness are sized with physics-based tools to achieve 2% power loss.

Preliminary Schematic



Backup

Orbit Trades: Notes

- Diagram, delta-v budget, and launch vehicle performance to each transfer orbit are provided in the charts below
  - Timelines for each option are representative only
  - *While the Delta-V budget includes the transfer and operational orbits, the eclipse times are only for the operational orbit*
  - *Momentum unloading Delta-V is a placeholder, with the same value being used for all options (can be revised after orbit downselect)*
- Orbit considerations include:
  - Delta-V requirements, thermal stability, environment, distance from Earth, launch vehicle / kick stage requirements, and science
  - *Assuming all options can fulfill the sky observing requirements*
  - *No* consideration given to launch windows

Delta-V Budget: SE-L2

Event/Maneuver	Start Date	MET (Days)	C3 (km2/s2)	Delta-V (m/s)	ACS Tax (%)	Margin (%)	Total (m/s)
Launch	1/1/30	0.0	-0.70				
Despin	1/1/30	0.0		5	0%	10%	5.5
Post-TTI correction	1/2/30	1.0		41	5%	10%	47.4
Additional correction for late launch	1/2/30	1.0		8	5%	10%	9.2
MCC-1	1/6/30	5.0		7.5	5%	10%	8.7
MCC-2	2/5/30	35.0		5	5%	10%	5.8
MCC-3 / Other (optional)	4/5/30	94.0		5	5%	10%	5.8
Stationkeeping (30 years)	7/4/30	184.0		72.9	5%	10%	84.2
Momentum unloading (30 years)	7/4/30	184.0		43.5	0%	10%	47.9
Disposal	1/1/50	7305.0		1	0%	10%	1.1
TOTALS				**188.9**			**215.5**

Values are based on JWST analyses. MET values are approximate.

ADVANCED CONCEPTS

Eclipse and Distance: SE-L2

Topic	Value	Units
Time to spacecraft separation	129	minutes
S/C separation in sunlight?	yes*	
Average eclipse	none	minutes
Longest eclipse	none	minutes
Average time between eclipses	na	minutes
Minimum time between eclipses	na	minutes
Max distance** in 1 yr	1,500,000	km
5 yr	1,500,000	km
10 yr	1,500,000	km
20 yr	1,500,000	km

* Trajectory can be designed such that separation occurs in sunlight, though this may impact launch windows.

** These values assume orbit maintenance maneuvers are completed (if required).

Eclipse times are for the target orbit, and do not include eclipses during the orbit transfer.

Delta-V Budget: LDRO

Event/Maneuver	Start Date	MET (Days)	C3 (km2/s2)	Delta-V (m/s)	ACS Tax (%)	Margin (%)	Total (m/s)
Launch	1/1/30	0.0	-1.80				
Despin	1/1/30	0.0		5	0%	10%	5.5
Post-TTI correction	1/2/30	1.0		41	5%	10%	47.4
MCC-1	1/3/30	2.0		50	5%	10%	57.8
Lunar Flyby	1/6/30	5.0		162	5%	10%	187.1
MCC-2	1/10/30	9.0		155	5%	10%	179.0
LDRO Insertion	1/17/30	16.0		3	5%	10%	3.5
Stationkeeping (30 years)	7/4/30	184.0		7.5	5%	10%	8.7
Momentum unloading (30 years)	7/4/30	184.0		43.5	0%	10%	47.9
Disposal	1/1/50	7305.0		10	0%	10%	11.0
TOTALS				**477.0**			**547.7**

Values are based on analysis.

ADVANCED CONCEPTS

Eclipse and Distance: LDRO

Topic	Value	Units
Time to spacecraft separation	129	minutes
S/C separation in sunlight?	yes*	
Average eclipse	211	minutes
Longest eclipse	706	minutes
Average time between eclipses	118516	minutes
Minimum time between eclipses	12640	minutes
Max distance** in 1 yr	500,000	km
5 yr	500,000	km
10 yr	500,000	km
20 yr	500,000	km

* Trajectory can be designed such that separation occurs in sunlight, though this may impact launch windows.

** These values assume orbit maintenance maneuvers are completed (if required).

Eclipse times are for the target orbit, and do not include eclipses during the orbit transfer.

Delta-V Budget: CTO

Event/Maneuver	Start Date	MET (Days)	C3 (km2/s2)	Delta-V (m/s)	ACS Tax (%)	Margin (%)	Total (m/s)
Launch	1/1/30	0.0	na				
Despin	1/1/30	0.0		5	0%	10%	5.5
Post-TTI correction	1/2/30	1.0		0	5%	10%	0.0
Additional correction for late launch	1/2/30	1.0		0	5%	10%	0.0
MCC-1	1/6/30	5.0		0	5%	10%	0.0
MCC-2	2/5/30	35.0		0	5%	10%	0.0
MCC-3 / Other (optional)	4/5/30	94.0		0	5%	10%	0.0
Stationkeeping (30 years)	7/4/30	184.0		0	5%	10%	0.0
Momentum unloading (30 years)	7/4/30	184.0		43.5	0%	10%	47.9
Disposal	1/1/50	7305.0		302	0%	10%	332.2
TOTALS				**350.5**			**385.6**

Eclipse and Distance: CTO

Topic	Value	Units
Time to spacecraft separation	407	minutes
S/C separation in sunlight?	yes*	
Average eclipse	54	minutes
Longest eclipse	265	minutes
Average time between eclipses	6743	minutes
Minimum time between eclipses	326	minutes
Max distance** in 1 yr	200,000	km
5 yr	200,000	km
10 yr	200,000	km
20 yr	200,000	km

*** Trajectory can be designed such that separation occurs in sunlight, though this may impact launch windows.**

**** These values assume orbit maintenance maneuvers are completed (if required).**

Eclipse times are for the target orbit, and do not include eclipses during the orbit transfer.

Delta-V Budget: TESS

Event/Maneuver	Start Date	MET (Days)	C3 (km2/s2)	Delta-V (m/s)	ACS Tax (%)	Margin (%)	Total (m/s)
Launch	1/1/30	0.0	na				
Despin	1/1/30	0.0		5	0%	10%	5.5
Launch vehicle error correction	1/2/30	1.0		25	5%	10%	28.9
Deterministic Maneuvers	1/14/30	1.0		150	5%	10%	173.3
Statistical Maneuvers	1/6/30	5.0		40	5%	10%	46.2
Other	2/5/30	35.0		0	5%	10%	0.0
Other	4/5/30	94.0		0	5%	10%	0.0
Stationkeeping (30 years)	7/4/30	184.0		0	5%	10%	0.0
Momentum unloading (30 years)	3/4/30	62.0		43.5	0%	10%	47.9
Disposal	1/1/50	7305.0		0	5%	10%	0.0
TOTALS				**263.5**			**301.7**

This Delta-V budget is for the TESS-type transfer, which includes a lunar gravity assist.

Since the lunar gravity assist is the mechanism for raising the perigee to the target value, and is a very large energy boost, it is probably not feasible to eliminate it. Having the launch vehicle place the satellite into the final orbit, or by having a kick stage perform the maneuver, is unlikely.

Eclipse and Distance: TESS

Topic	Value	Units
Time to spacecraft separation	129	minutes
S/C separation in sunlight?	yes*	
Average eclipse	107	minutes
Longest eclipse	272	minutes
Average time between eclipses	217	days
Minimum time between eclipses	53	days
Max distance** in 1 yr	500000	km
5 yr	500000	km
10 yr	500000	km
20 yr	500000	km

*** Trajectory will most likely be such that separation occurs in sunlight.**
**** No orbit maintenance is required to maintain these bounds.**

Eclipse times are for the target orbit, and do not include eclipses during the orbit transfer.

Delta-V Budget: DAO

Event/Maneuver	Start Date	MET (Days)	C3 (km2/s2)	Delta-V (m/s)	ACS Tax (%)	Margin (%)	Total (m/s)
Launch	1/1/30	0.0	0.61				
Despin	1/1/30	0.0		5	0%	10%	5.5
Post-TTI correction	1/2/30	1.0		0	5%	10%	0.0
Additional correction for late launch	1/2/30	1.0		0	5%	10%	0.0
MCC-1	1/6/30	5.0		0	5%	10%	0.0
MCC-2	2/5/30	35.0		0	5%	10%	0.0
MCC-3 / Other (optional)	4/5/30	94.0		0	5%	10%	0.0
Stationkeeping (30 years)	7/4/30	184.0		0	5%	10%	0.0
Momentum unloading (30 years)	7/4/30	184.0		43.5	0%	10%	47.9
Disposal	1/1/50	7305.0		0	0%	10%	0.0
TOTALS				**48.5**			**53.4**

Eclipse and Distance: DAO

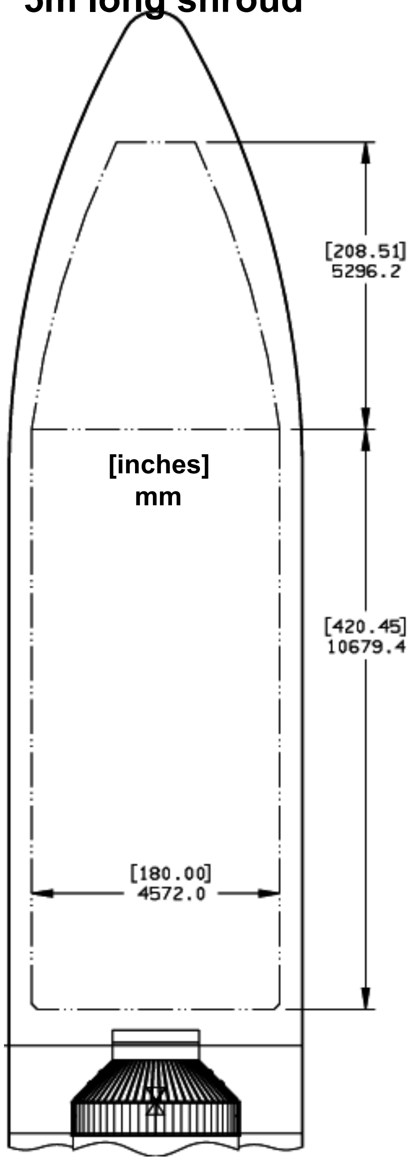
Topic	Value	Units
Time to spacecraft separation	129	minutes
S/C separation in sunlight?	yes*	
Average eclipse	none	minutes
Longest eclipse	none	minutes
Average time between eclipses	na	minutes
Minimum time between eclipses	na	minutes
Max distance** in 1 yr	0.1	AU
5 yr	0.6	AU
10 yr	1.1	AU
20 yr	1.8	AU

*** Trajectory will most likely be such that separation occurs in sunlight.**
**** A higher launch C3 can perhaps reduce these values. Analysis is pending.**

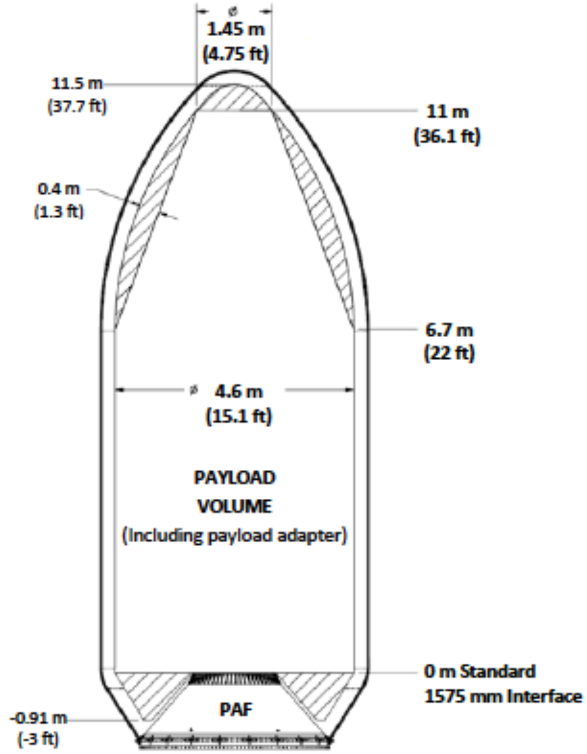
Eclipse times are for the target orbit, and do not include eclipses during the orbit transfer.

Launch Vehicle: Representative Shrouds (not to scale)

Altas V
5m long shroud



Falcon 9 / Heavy



Delta-IV Heavy